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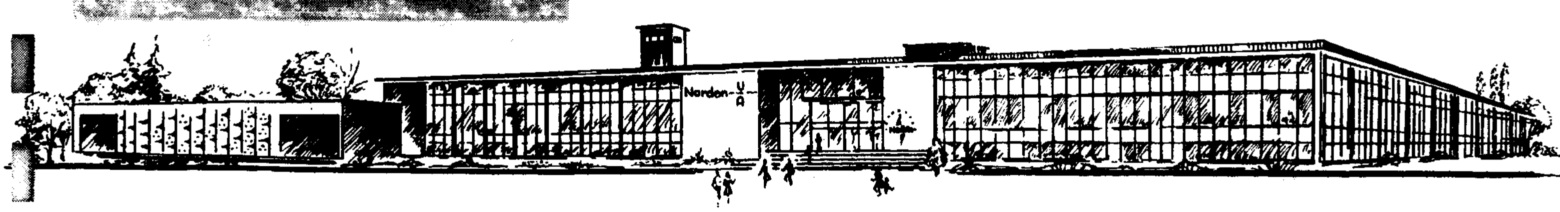

NORWALK, CONNECTICUT

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Serial No. 32

FINAL ENGINEERING REPORT
for a
STUDY OF
DISPLAY INTEGRATION
for
HYPERSONIC RESEARCH VEHICLES

1141 R 0004

15 MAY TO 15 NOVEMBER 1963

Prepared for:

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Space Administration
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Contract NASw-675

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FOREWORD

This is the final engineering report pertaining to the work performed by Norden Division of United Aircraft Corporation under NASA Contract NASw-675. The report covers the period from 16 May 1963 to 16 November 1963.

The report summarizes the results of a 6-month study program for the investigation of cockpit display integration based on the mission requirements of the X-15 research vehicle. The program was conducted by the Video Systems Engineering Group of Norden under the technical supervision of Mr. Roger Winblade, Display and Guidance Section, NASA Flight Research Center, Edwards, California.

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ABSTRACT

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The mission profile of the X-15 research vehicle is analyzed in terms of pilot display requirements and vehicle characteristics. Present methods of flight information display are discussed.

Design criteria for integrated display systems are presented and a display format based on contact analog techniques is developed.

Pilot control tasks for a selected mission are analyzed in detail and application of the integrated display is described. A mechanization of a complete display system is described in terms of performance capabilities and physical characteristics, and is related to present display methods.

Simulator evaluation programs involving integrated display systems of a similar nature are outlined. Recommendations are included for follow-on programs involving simulator experimentation, airborne evaluation, and display improvement.

AUTHOR

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

1. INTRODUCTION

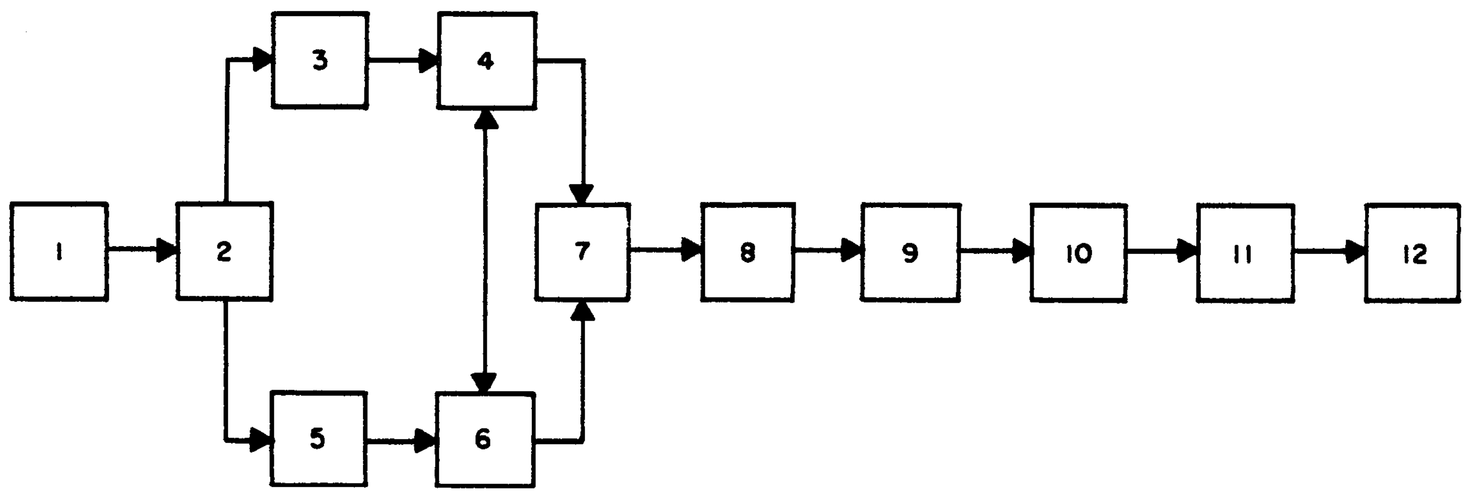
As the role of the pilot gains importance in the control of space vehicles, it is becoming evident that improved methods of information display are required. The many rapidly changing parameters involved in these flights place new demands upon the designer to produce an effective integrated instrument display which is usable throughout the entire mission. Critical decisions must be made based on several variables occurring simultaneously. A method of display integration is needed which will reduce both the number of separate indicators used and the time required to make critical decisions.

It is the purpose of this program to study means of extending pilot vehicle performance through the use of advanced display techniques. The X-15 research aircraft was selected to serve as the basis of the investigation, since it is the only operational manned aircraft capable of bridging the gap between high altitude aerodynamic flight and ballistic space flight.

1.1 Method of Approach

The procedure used in this investigation follows the sequence shown in Figure 1-1. The steps shown are based on the methods of analysis outlined in Section 15 , reference 1.

First, the total system requirements are defined in terms of definition of the vehicle and the mission profile. The pilot tasks relating to the mission are then outlined and the parameters required for accomplishment of these tasks are specified. Present methods of displaying these parameters are analyzed in detail. An integrated display using electronic analog techniques with pictorial presentation is developed and analyzed in terms of the following:



1. X-15 VEHICLE AND PROGRAM FAMILIARIZATION
2. MISSION PROFILE ANALYSIS
3. FLIGHT PARAMETER AND SENSOR STUDY
4. ANALYSIS OF PRESENT DISPLAY METHODS
5. PILOT TASK ANALYSIS
6. PILOT TASK-DISPLAY PARAMETER MATRIX
7. STUDY OF DISPLAY INTEGRATION METHODS
8. RECOMMENDED DISPLAY CONTENT AND FORMAT
9. TECHNIOUES OF MECHANIZATION
10. SIZE, ACCURACY, RELIABILITY OF SYSTEM
11. RECOMMENDATIONS FOR FOLLOW-ON PROGRAM
12. FINAL REPORT

Figure 1-1. Study Procedure

a. Number of displayed variables

b. Readability

c. Pilot attention requirements

d. Display flexibility

e. Mechanization

Conclusions and recommendations for further investigation are also presented.

2. VEHICLE DESCRIPTION

The X-15 Research Vehicle (Figure 2-1) is a rocket-powered single-place aircraft designed to perform basic research in the problems of manned flight at extremely high altitudes. Three aircraft were built for the program and have been in use for more than seven years. One of these is currently undergoing modifications for extended performance capabilities. The other two are operational aircraft and are utilized as test vehicles for numerous space and near-space experimentation programs. A sampling of these programs is given in Table 2-1.

The physical and operational characteristics of the X-15 are listed in Appendix A. The aircraft is powered by a liquid-propellant rocket engine using anhydrous ammonia and liquid oxygen. The engine is designed to operate for approximately 1.5 minutes at maximum thrust. The propellant tanks occupy most of the fuselage area to the rear of the pilot compartment. The engine itself occupies the area within the tail section. The main wing has a 25.5-degree sweep with hydraulically operated flaps. The horizontal stabilizer has a 15-degree cathedral. The two sections move in unison for pitch control, differentially for roll control, and in compound for pitch roll control. The upper and lower vertical stabilizers are in two sections: a movable outer span for yaw control and a fixed section adjacent to the fuselage. The lower movable section (ventral) is jettisoned for landing. (Flights have been made without this section for stability investigations.) Each fixed section of the vertical stabilizers incorporates a split-flap speed brake. These are used during both powered and unpowered flight to control speed and acceleration (refer to Section 15, reference 2).

For changes in aircraft attitude relative to flight trajectory at altitudes where aerodynamic controls are relatively ineffective, a ballistic control system (BCS) is provided, wherein

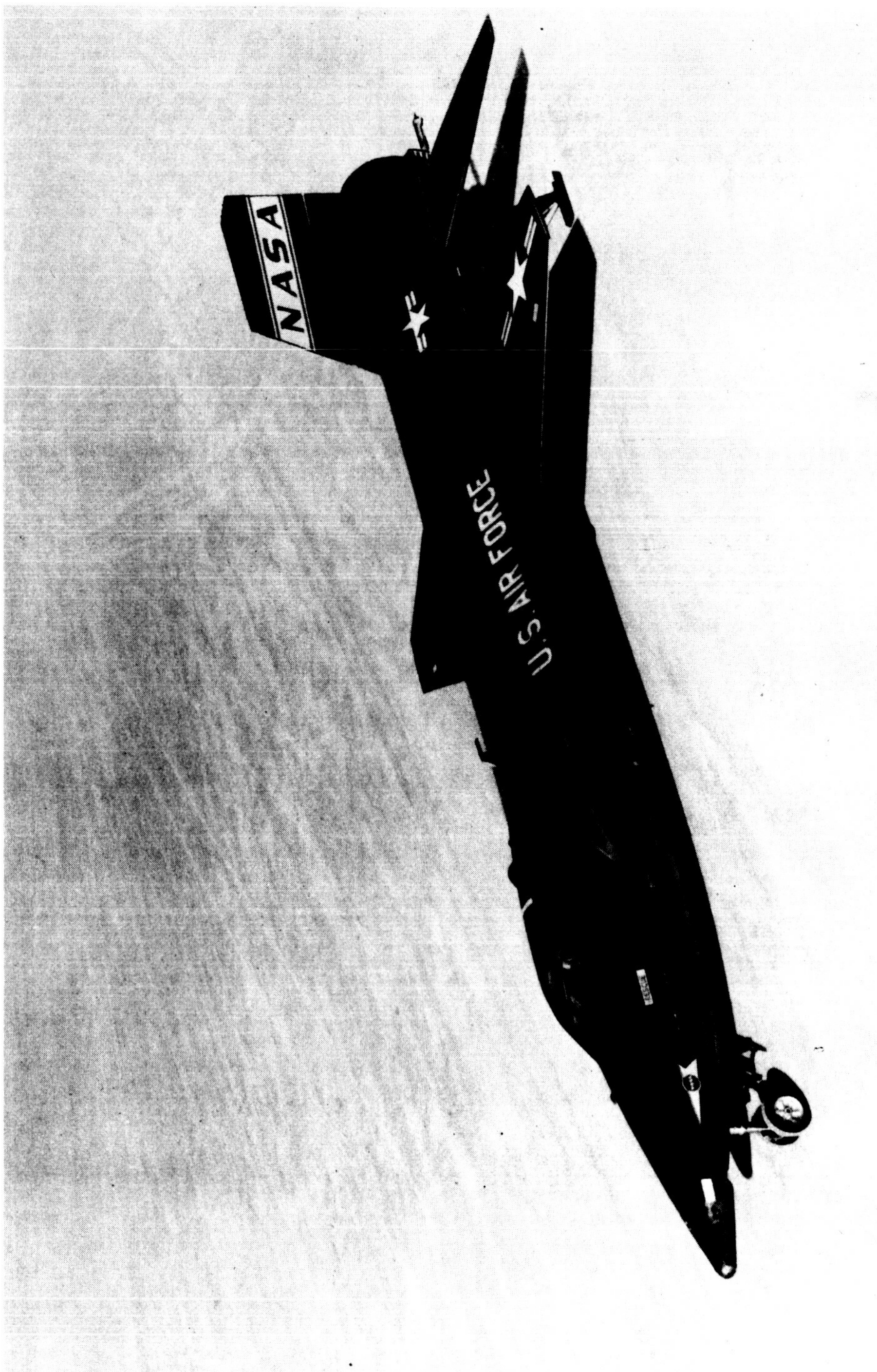


Figure 2-1. X-15 Research Vehicle

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Table 2-1. Current X-15 Experimentation Program

NUMBER	EXPERIMENT	MISSION
1	Ultraviolet stellar photography	High altitude
2	Ultraviolet exhaust plume characteristics	Above 25 miles
3	Horizon definition	Above 40 miles
4	Optical-degradation measurements	Varied
5	Detachable high-temperature leading edges	High heating
6	Infrared exhaust signature	100,000 feet to 130,000 feet
7	High-temperature windows	Varied
8	Atmospheric-density measurements	Above 125,000 feet
9	Micrometeorite collection	Above 150,000 feet
10	Advanced integrated-data systems and energy management	Varied
11	Vapor-cycle cooling	Long duration zero g

the metered release of hydrogen peroxide through small rockets in the nose and wing cause the aircraft to accelerate about each axis as required.

A conventional stability augmentation system is incorporated in two of the X-15 vehicles. The third is fitted with an adaptive control system, which is capable of changing the gain of aircraft control and stability loops to compensate for the wide range of environmental conditions encountered.

The aircraft is not designed for ground takeoff, but is air-launched by a B-52 mother ship. The landing gear consists of a dual-wheel nose gear and two main landing skids in the rear. The gear is lowered in flight by gravity and air loads. Two auxiliary power units (APU's) drive the hydraulic pumps and a-c electrical generators. The approximate launch gross weight including full internal load and pilot is 32,900 pounds. However, this can vary depending on the type of instrumentation carried.

A full six-degree-of-freedom fixed-base simulator has been in use since the beginning of the program for pilot training and calculation of vehicle performance. The simulator has proven to be an accurate means of predicting flight characteristics and stability problems.

3. MISSION PROFILE

The X-15 was designed for a maximum altitude of 250,000 feet and velocity of 6600 feet per second (4500 mph). Figure 3-1 shows the vehicle performance envelope in the typical altitude-velocity format. The solid curve shows the predicted altitude and velocity for the YLR-99 engine. The lower dashed line, included for reference purposes, is for constant dynamic pressure of 1500 psf, although the design limit is 2500 psf. The crosshatched area shows that a maximum altitude of 354,200 feet* and a maximum velocity of 6020 feet per second have been attained. Although additional performance potential remains to be explored, attainment of altitudes of the order of 400,000 feet or greater (shaded area) may be limited because of entry problems shown in Figure 3-1 (refer to Section 15, reference 3).

The particular flight profile flown for a given mission depends entirely on the purpose of the mission. Thus, a detailed description of each mission will not be attempted here. Rather, these missions can be grouped into the following two major categories:

a. Altitude missions (involving flights beyond the sensible atmosphere)

b. Speed missions (involving flights completely within the atmosphere)

Altitude missions include flights for purposes such as pilot control studies, spacecraft instrumentation tests, investigation of reentry dynamics, near-space physical and astronomical studies, and zero g environment studies. Speed missions include high-temperature structural tests, high-speed stability investigations, and evaluation of special instrumentation (refer to Table 2-1).

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* record established August 22, 1963

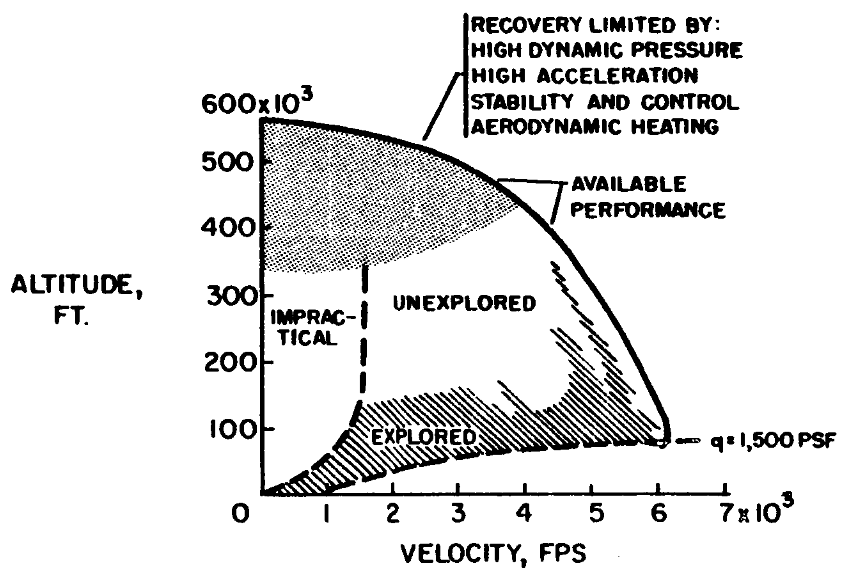

Figure 3-1. X-15 Performance Envelope.

Altitude profiles (Figure 3-2a) are characterized by a high pitch-angle climbout followed by a sustained period of ballistic (zero g) flight and reentry into the atmosphere. The relative severity of the reentry phase is a function of the peak altitude and peak velocity attained, therefore, the control accuracy requirements during the powered portion of such flights are extremely critical. Since pitch angle variations of one degree can result in peak altitude errors of several thousand feet, the pilot must be continually advised of his position with respect to the planned profile, so he may correct his position before engine shutdown. Once shutdown has occurred, the pilot usually has no control over his trajectory. Engine cutoff must be controlled carefully as well as flight path to minimize trajectory dispersion.

The boost, ballistic, and reentry phases of the X-15 altitude profile are closely related to the corresponding phases of manned orbital flight. The principal difference between boost phases is in the launch attitude (X-15: horizontal, orbital vehicle: vertical). However, problems concerning pilot authority, trajectory accuracy, and system reliability are similar. Studies concerning pilot participation in boost guidance are being conducted using X-15 test data. Vehicle attitude control problems during ballistic flight are also similar in nature, although X-15 attitude changes are restricted by the relatively short duration of this flight phase. In spite of this limitation, many types of space vehicle systems have been evaluated in X-15 flights.

The reentry phase of the X-15 altitude profile can be compared to the latter reentry phases of maneuverable reentry vehicles, where speed, altitude, and glide-range considerations are identical in nature. A brief discussion of the display problems for such vehicles will be undertaken in Appendix E of this report.

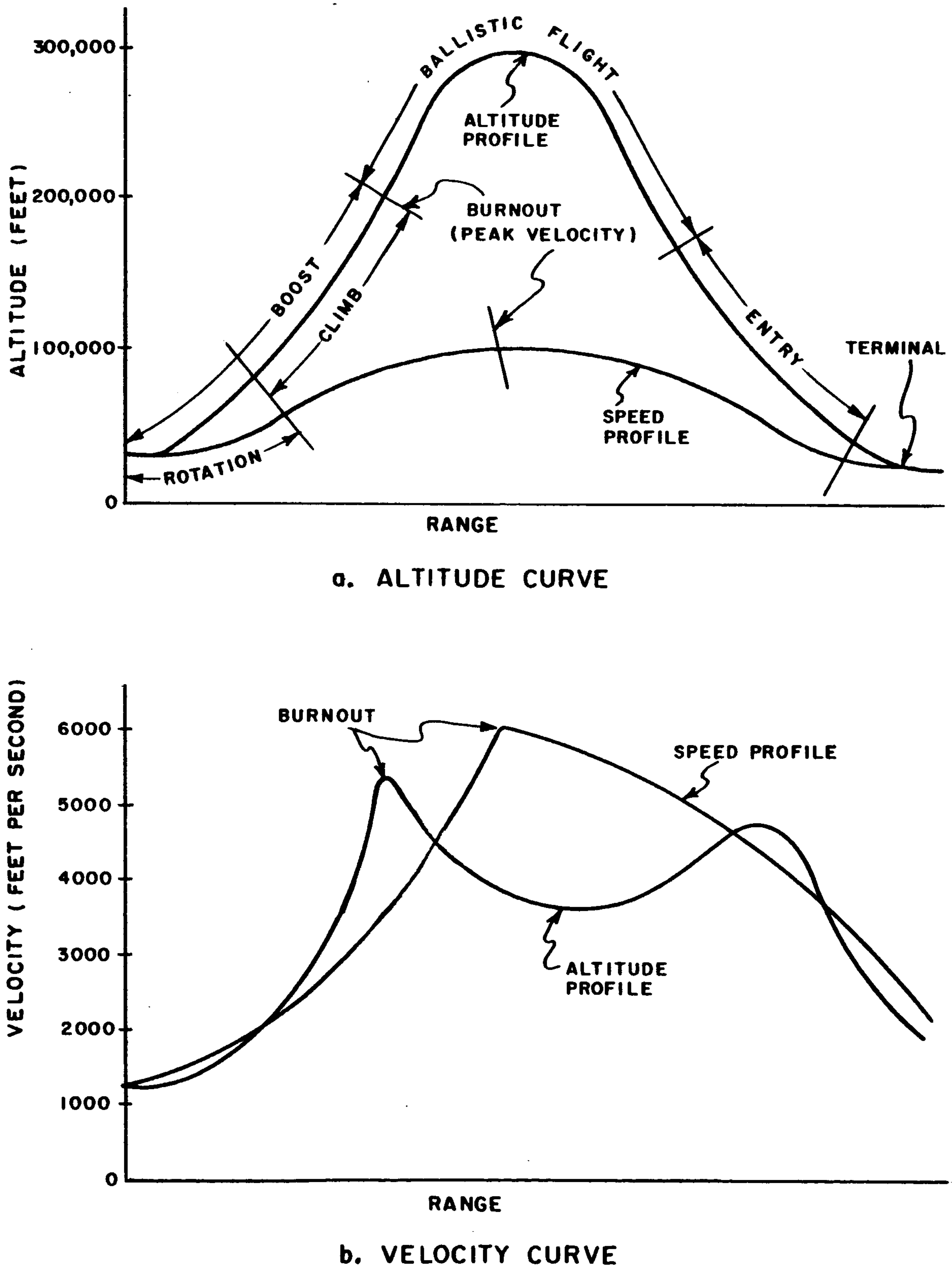

Figure 3-2. X-15 Flight Profiles

Speed profiles (Figure 3-2b) require a relatively shallow flight path within the atmosphere. Zero-g flight is maintained by the pilot, using acceleration and angle of attack cues.

When maximum aerodynamic heating is the flight objective, the pilot must maintain a combination of constant velocity and constant altitude (calculated to produce constant dynamic pressure) until engine burnout. The glide phase of such flights is often used for stability and instrumentation tests.

All missions are initiated with an air drop from a B-52 mother ship at approximately 45,000 feet and subsonic speed. Engine ignition occurs immediately, followed by a rotation maneuver to the desired pitch angle for climbout. Once the correct pitch angle is established, the flight progresses according to a flight plan calculated to produce the desired profile. Landings are conducted in accordance with an established pattern which is essentially identical for all flights.

4. PILOT TASKS

A compilation of representative pilot tasks for many types of missions is given in Table 4-1. This table lists only tasks pertaining to accomplishment of a particular mission. These must be performed in addition to the normal, or recurring tasks, required to operate the aircraft in all other respects, such as maintaining trim, attitude, heading, and voice communications.

Specific information for each mission is transmitted to the pilots in the form of flight request sheets, giving a breakdown of all flight control requirements on a second-by-second basis from the time of launch. Each entry gives the altitude, velocity, angle-of-attack, dynamic pressure, and special pilot instructions corresponding to a particular time. Emergency and abort information is also included. (Refer to Appendix C.)

The present method of operation requires that the pilot memorize this information prior to each flight. A basic requirement of an integrated display would be the capability of presenting this type of command information during flight.

A detailed breakdown of pilot tasks for a selected altitude mission is given in Section 9, Figure 9-2.

Table 4-1. Representative Pilot Tasks

FLIGHT PHASE	TASK
Launch	a. Drop b. Ignite engine c. Hold 2 degree right aileron
Rotation	a. Maintain constant heading b. Maintain constant 2 g until given pitch angle
Climb	a. Maintain constant pitch angle; heading b. Reduce thrust at given time c. Shutdown engine at given time; velocity d. Pushover to zero g at given time; speed; altitude; or rate-of-climb e. Monitor maximum velocity
Ballistic	a. Maintain zero g b. Maintain constant vertical speed c. Extend speed brakes d. Maintain level flight e. Maintain maximum 8 degree pitch and roll error f. Roll in and release given bank angle g. Fly proportional to bank angle; roll rate h. Monitor peak altitude
Entry	a. Establish given angle of attack b. Perform airleron deflection pulse c. Perform rudder deflection pulse d. Monitor dynamic pressure e. Monitor angle of attack

Table 4-1. Representative Pilot Tasks (Continued)

FLIGHT PHASE	TASK
Entry (cont)	f. Retract speed brakes g. Hold constant g pullout h. Monitor vertical speed i. Make level flight recovery
Terminal	a. At key checkpoints: Establish proper altitude, airspeed, ground position, turn rate, and rate-of-descent b. Jettison vertical fin at given altitude c. Before Flareout: Establish proper airspeed, height above runway, and sink-rate d. Lower flaps; gear e. Touchdown at given point on runway

5. FLIGHT PARAMETERS

The parameters normally used in controlling the X-15 are defined in Table 5-1, with corresponding symbols shown. Methods of vehicle control for selected parameters are discussed in Section 7.

Important considerations in the design of integrated flight displays are the ranges of variation encountered in the flight parameters during actual operation and the maximum rates of change of these parameters. Table 5-2 lists these characteristics for the X-15 flight parameters, indicating the sensors which furnish the input information.

It is apparent that the rates of change associated with attitude parameters are of such magnitude as to require display devices capable of following large changes with minimum time lag and overshoot. The extremely wide ranges of variation associated with altitude and speed parameters require the additional capabilities of large dynamic range with high sensitivity. A combined display capable of handling both groups of parameters must, therefore, possess the opposing characteristics of fast response, large dynamic range, and high sensitivity with readability. These flight parameters are analyzed in detail for a selected flight profile in Section 9.

Table 5-1. Parameter Definitions

PARAMETER	SYMBOL	DEFINITION	UNIT
Acceleration (normal)	G, g, a_N	Acceleration acting along vehicle normal axis	g units
Airspeed	IAS	Airspeed as indicated by the pressure-type airspeed indicator	knots
Altitude (pressure)	H_p	Barometric altitude above sea level as indicated by the pressure altimeter	feet
Angle-of-Attack	α	The angle between the relative wind and the chord of the wing	degrees
Angle-of-Sideslip	β	The angle between the relative wind and the plane formed by the aircraft vertical and longitudinal axis	degrees
Course Angle	Ψ_c	The angle in the plane of the earth's surface between the aircraft flight path and true north	degrees
Dynamic Pressure	q	$q = \frac{1}{2} \rho V^2$ where ρ = mass density of air V = vehicle velocity	pounds per square foot
Flight Path Angle	γ	The angle between the velocity vector and a true horizontal in the vertical plane	degrees
Heading Angle	Ψ_h	The angle between the aircraft fore-aft axis and true north	degrees

Table 5-1. Parameter Definitions (continued)

PARAMETER	SYMBOL	DEFINITION	UNIT
Heading Error	$\Delta\Psi_h$	The angular difference between the aircraft heading and a command heading angle	degrees
Inertial Climb	$\dot{h}$, V_v	Aircraft vertical velocity as measured by the inertial flight data system	feet per second
Inertial Height	H, h	Aircraft vertical displacement with respect to a preselected reference height as measured by the inertial flight data system	feet
Inertial Speed	V	Aircraft total velocity with respect to the earth's surface, as measured by the inertial flight data system	feet per second
Pitch Angle	θ	The angle between the aircraft longitudinal axis and a true horizontal plane	degrees
Pitch Error	θ_e	The angular difference between the aircraft pitch angle and a command pitch angle.	degrees
Peak Altitude	h_{max}	The maximum altitude attained by the aircraft in a given flight	feet
Roll Angle	ϕ	The angle between the lateral axis and a true horizontal in the plane formed by the lateral and normal axes	degrees

Table 5-1. Parameter Definitions (continued)

PARAMETER	SYMBOL	DEFINITION	UNIT
Roll Rate	p	The time derivative of roll angle	degrees per second
Time From Launch	t	Flight time relative to the opening of the main propellant valves	seconds
North-South Velocity	V_{NS}	Aircraft velocity relative to the earth's surface in the direction of true north as measured by the inertial flight data system	feet per second
East-West Velocity	V_{EW}	Inertial velocity component in the horizontal plane perpendicular to North-South velocity	feet per second

NOTE: For angular relationships, refer to Appendix F

TABLE 5-2. Characteristics of X-15 Flight Parameters

PARAMETER	TYPICAL RANGE OF VARIATION	MAXIMUM RATE OF CHANGE	SENSOR
Barometric Altitude	0 to 60,000 feet	------------	Pitot-Static System
Inertial Height	45000to 500,000 feet	$\pm$4000 feet per second	Platform
Inertial Speed	700 to 7,000 feet per second	120 feet per second squared	Platform
Airspeed	0 to 450 knots	------------	Pitot-Static System
Mach Number	Mach 0.2 to Mach 2.5	------------	Ball Nose
Vertical Speed	$\pm$4000 feet per second	------------	Platform
Dynamic Pressure	0 to 2000 pounds per square feet	75 pounds per square foot per second	Ball Nose (total pressure)
Angle-of-Attack	-10 to +35 degrees	12 degrees per second	Ball Nose
Angle-of-Sideslip	$\pm$10 degrees	10 degrees per second	Ball Nose
Pitch	$\pm$60 degrees	12 degrees per second	Platform
Heading	360 degrees	10 degrees per second	Platform
Roll	$\pm$120 degrees	30 degrees per second	Platform
Time	0 to 800 seconds	------------	Stopwatch

Table 5-2. Characteristics of X-15 Flight Parameters (continued)

PARAMETER	TYPICAL RANGE OF VARIATION	MAXIMUM RATE OF CHANGE	SENSOR
Acceleration			
Longitudinal	-2 g to +4 g	-------------	Accelerometer
Lateral	-1 g to +1 g	-------------	Accelerometer
Normal	0 g to +6 g	-------------	Accelerometer
Ground Track Distance	325 miles	1 mile per second	Platform

6. FLIGHT SENSORS

Two primary sensors provide the input data for the flight instruments. These include:

a. Inertial Flight Data System (stable platform)
b. Hypersonic Flow Direction Sensor (ball nose)

They are discussed in detail in the following paragraphs.

6.1 Inertial Flight Data System *

Input information for displaying vehicle velocity, height, rate-of-climb, and attitude is provided by the inertial flight data system. A functional diagram of this system is shown in Figure 6-1. The two major system components are the stabilizer, which contains the sensory elements, and the computer, which processes data from the sensors to furnish the necessary outputs.

Initial conditions necessary for proper alignment of the system prior to launch are provided from a control panel in the B-52 mother ship. Initial velocity is determined by means of Doppler radar and is transformed into along-range and across-range components in the control panel. Rate-of-climb information is introduced from a barometric rate-of-climb sensor, and heading is derived from an N-1 compass system. Initial position and altitude are manually inserted.

A block diagram of the on-board inertial system is shown in Figure 6-2. The stabilizer, or stable platform, is maintained normal to the local gravity vertical under all possible conditions of attitude and acceleration by means of three integrating gyros on a mount having four separate gimbals. The sequence of rotation from the case inwards is roll, pitch, inner roll, and azimuth. The pitch, inner roll, and azimuth axes are maintained mutually

* Section 15: References 7,8.

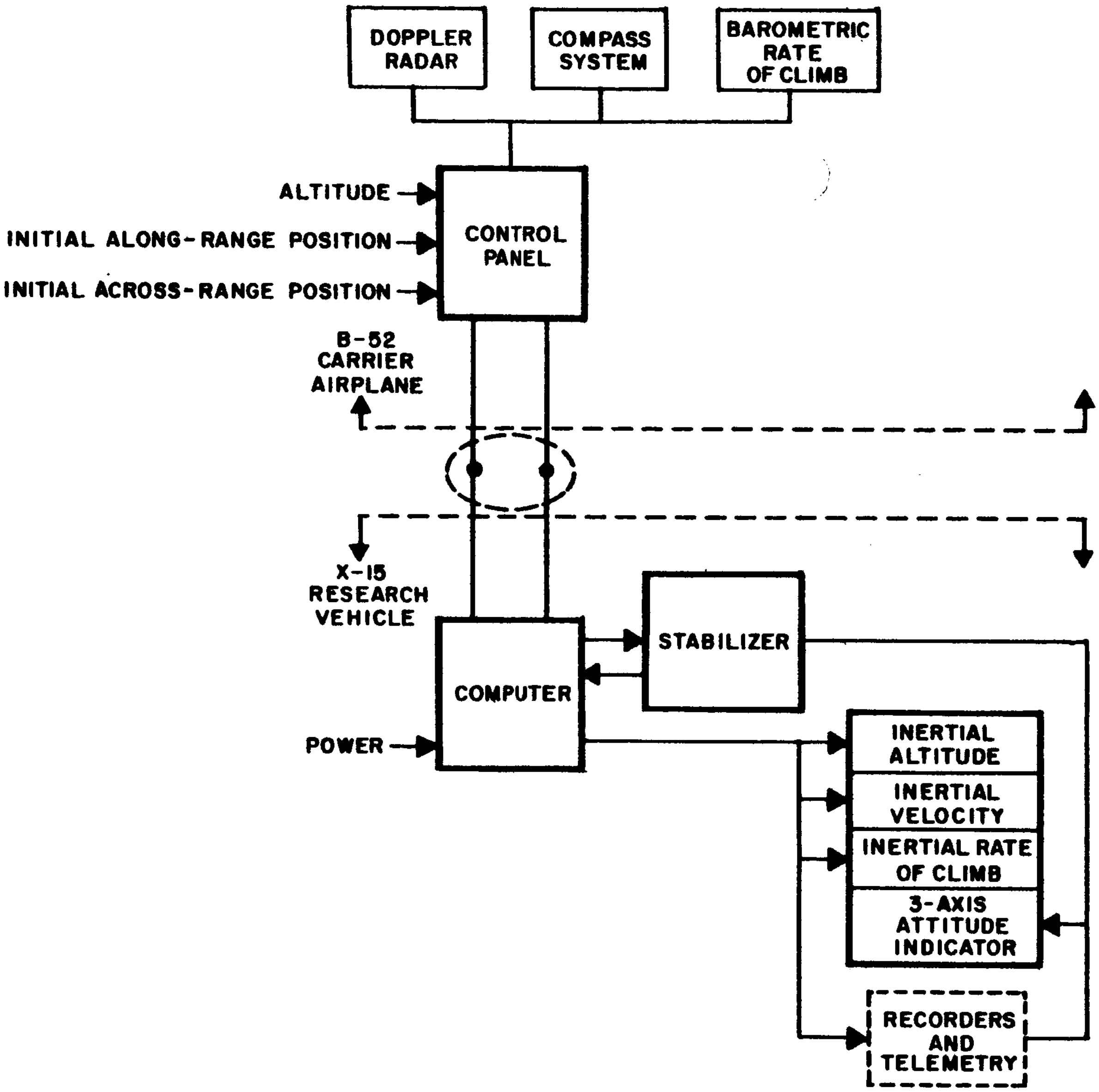

Figure 6-1. Inertial Flight Data System, Functional Diagram

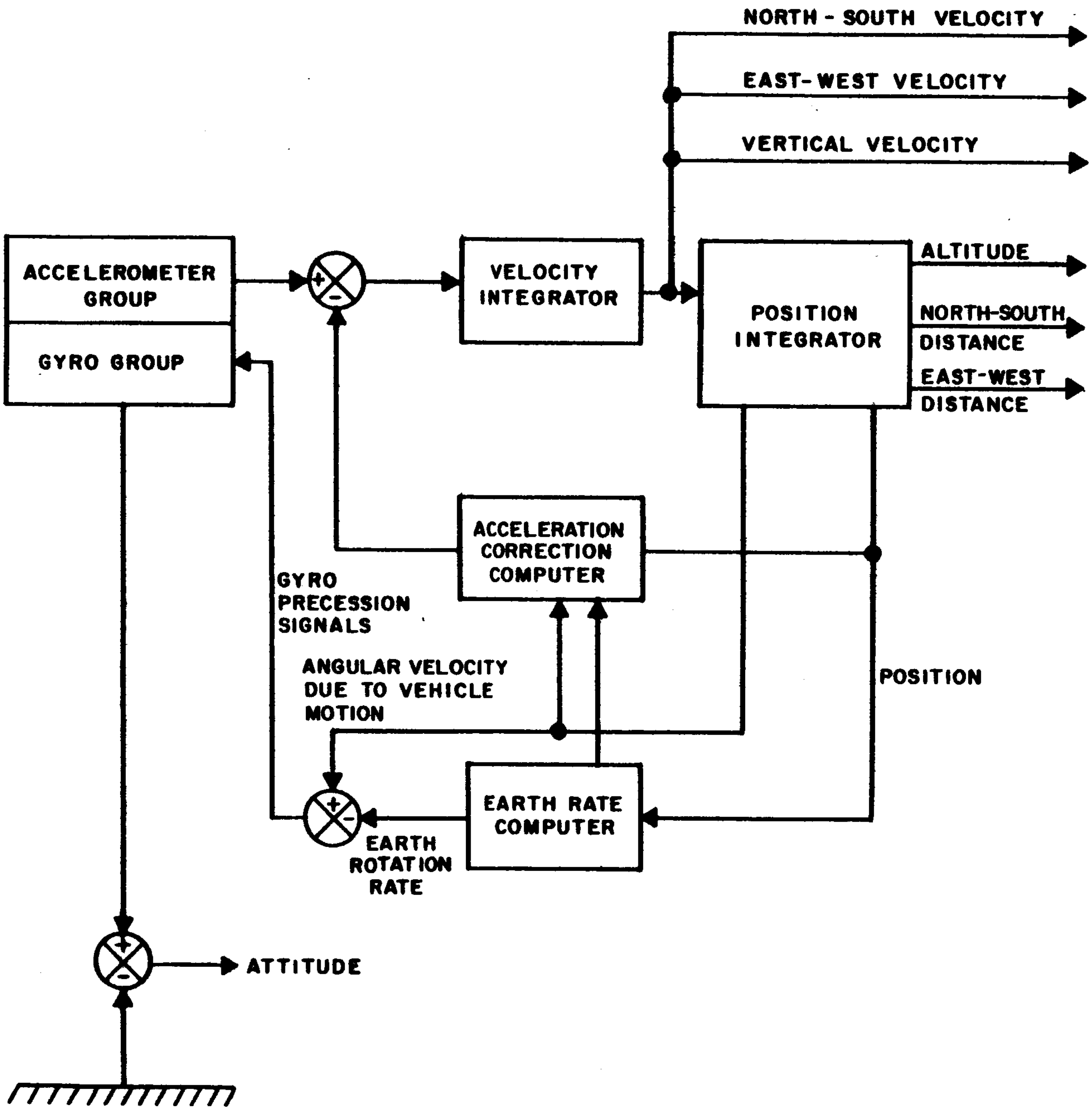

Figure 6-2. Inertial System, Block Diagram

perpendicular, while the outer-roll axis is gimbaled with respect to the aircraft fore-aft axis. This redundant roll gimbal is necessary to allow unlimited maneuverability about any axis without the possibility of gimbal lock or loss of verticality. The servo amplifiers for each axis are mounted directly on the gimbal being driven.

Three linear accelerometers are mounted on the platform and act as the primary sensing elements for velocity and position information. These have an accuracy in the order of 10^{-4} g.

The computer processes the accelerometer outputs to derive the variables to be displayed in the proper form. As a necessary part of this function, it also furnishes earth-rate and acceleration correction signals to precess the gyros. Earth-rate corrections are necessitated by the requirement for displaying velocity with respect to the earth's surface, since the platform senses only velocity with respect to inertial space. Acceleration corrections compensate for errors introduced by Coriolis and Centripetal accelerations, as well as errors resulting from variations in mass attraction with altitude due to gravity gradient.

The computer outputs are furnished to the displays both as synchro signals and as potentiometer resistances. These interface data are given in Appendix B.

6.1.1 Accuracy of Inertial Data

The accuracy characteristics of the inertial flight data system are shown in Figure 6-3. Velocity information is sufficiently accurate to be used for flight control over most of the flight. However, altitude error increases with time until, near the end of the flight, errors result of more than ±10,000 feet. This precludes use of inertial height information over all but the initial

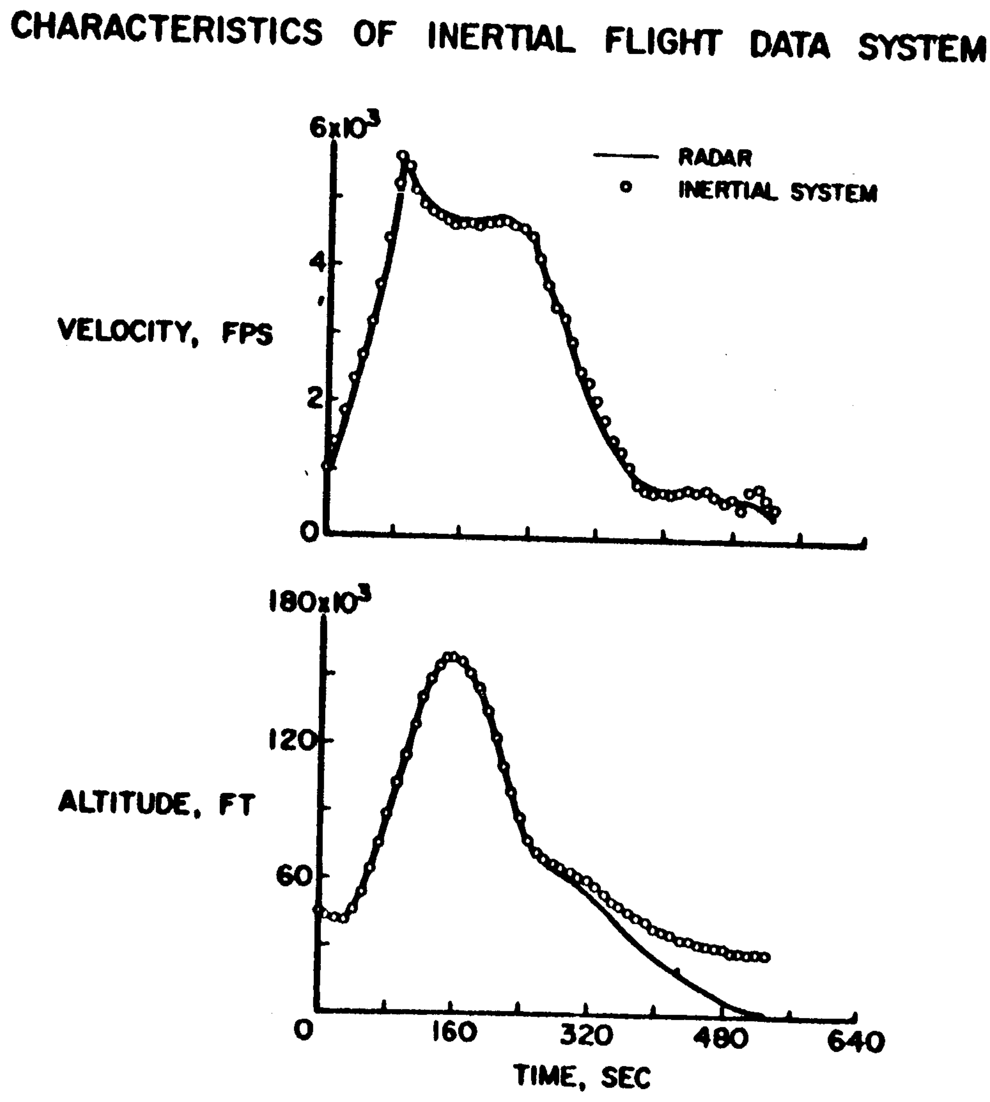

Figure 6-3. Inertial Flight Data System Accuracy

phases of flight. For this reason, altitude cues from ground radar data must be furnished to the pilot by voice communications in later flight phases.

Attitude information is sufficiently accurate throughout the entire flight, since it involves no integrations from platform data.

A further discussion of inertial data accuracy is given in paragraph 7.6.

6.2 Hypersonic Flow Direction Sensor

The hypersonic flow direction sensor (Figure 6-4) provides inputs for the display of the following parameters:

a. Angle of attack
b. Yaw angle
c. Dynamic pressure

A block diagram of this system is shown in Figure 6-5. The sensor itself consists of a 6.5-inch diameter servoactuated sphere, free to rotate about the vertical and horizontal axes. As the aircraft encounters a sideslip condition or change in angle-of-attack, the hydraulic servo system turns the ball into the relative wind. The difference between aircraft heading and relative wind is then transmitted through synchros to the cockpit indicators (Section 15, reference 4).

Two separate but identical servo systems operate the sphere in its two axes. Only the vertical axis is shown in Figure 6-5. An out-of-null wind condition causes a pressure differential between the upper and lower orifices. This is sensed by a transducer, whose output is proportional to the error. This electrical signal is then amplified by a variable gain amplifier, after which

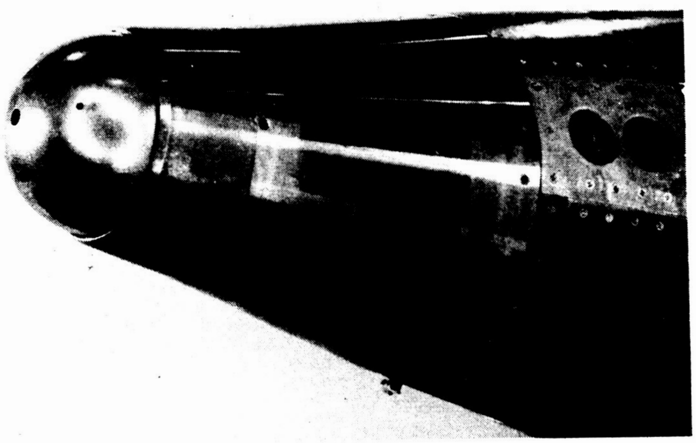

Figure 6-4. Hypersonic Flow Direction Sensor (Ball Nose)

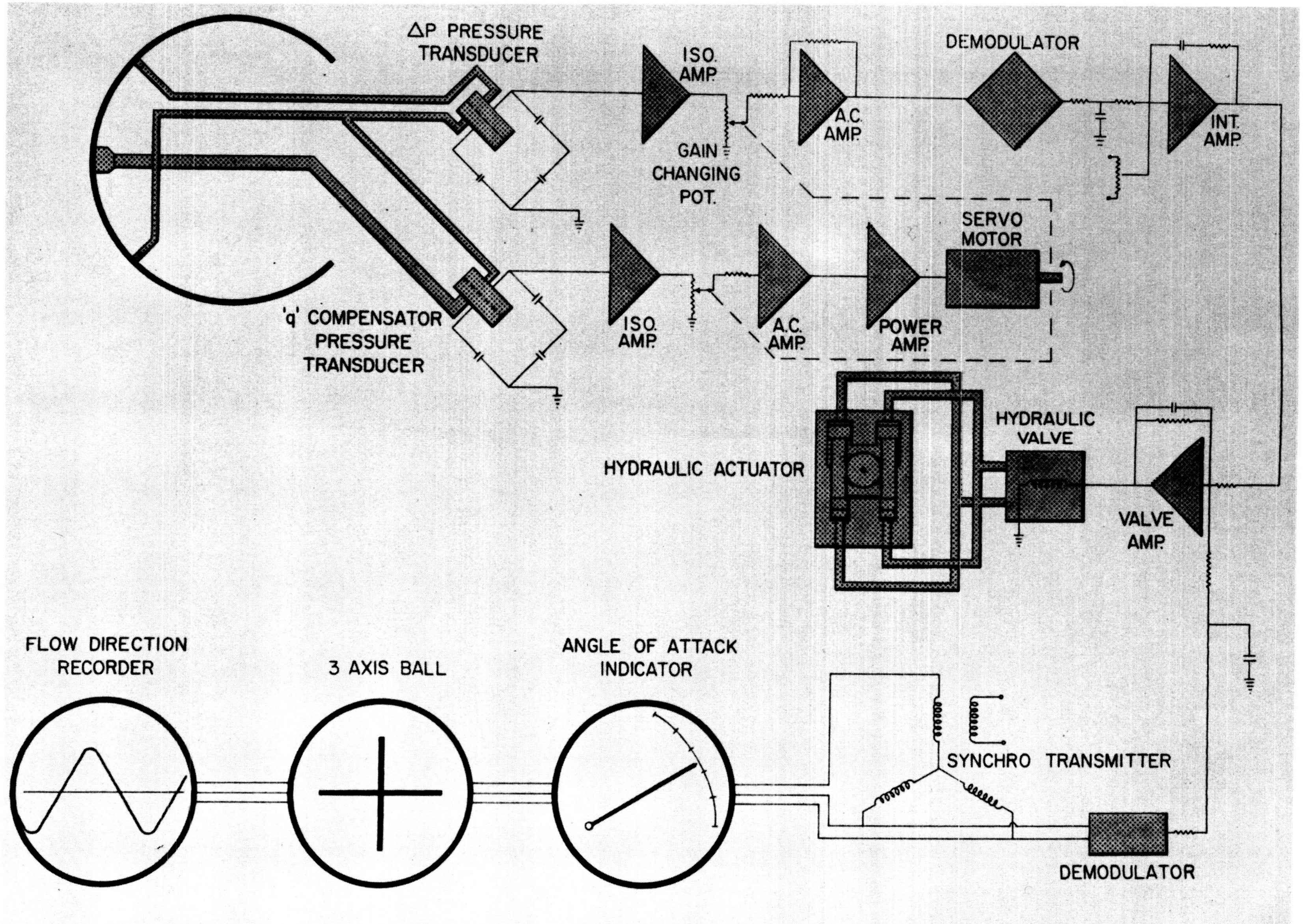

Figure 6-5. Ball Nose Block Diagram

it is demodulated and used to actuate the hydraulic turning mechanism. The gain of the amplifier is varied in accordance with the total dynamic pressure (q). This ensures stable operation over the total range of dynamic pressure from 15 to 2500 pounds per square foot. Since the temperatures at this point approach 1200°F, the sphere is cooled with liquid nitrogen from the cabin pressurization source.

The accuracy characteristics of the flow direction sensor are shown in Figure 6-6. The error is relatively constant at 0.6 degree from 1000 pounds per square foot to less than 100 pounds per square foot. As pressure decreases, the error increases and reaches a value of 2.2 degrees at one pound per square foot.

It should be noted that a minimum pressure of 25 pounds per square foot is required for aircraft control effectiveness and 100 pounds per square foot for normal flight.

Sensor frequency response is flat to 2 cycles per second and usable to 5 cycles per second, after which the phase lag becomes appreciable.

6.3 Pitot-Static System

The necessary pressure inputs for the pressure altimeter and airspeed-Mach are provided by the pitot probe and flush static system. The pitot probe furnishes stagnation pressure and is mounted directly ahead of the canopy, and 70 inches to the rear of the nose (Figure 2-1). Static pressure is supplied by ports on each side of the fuselage and forward of the cockpit area.

A detailed discussion of the calibration and accuracy of these sensors is given in Section 15 , reference 5.

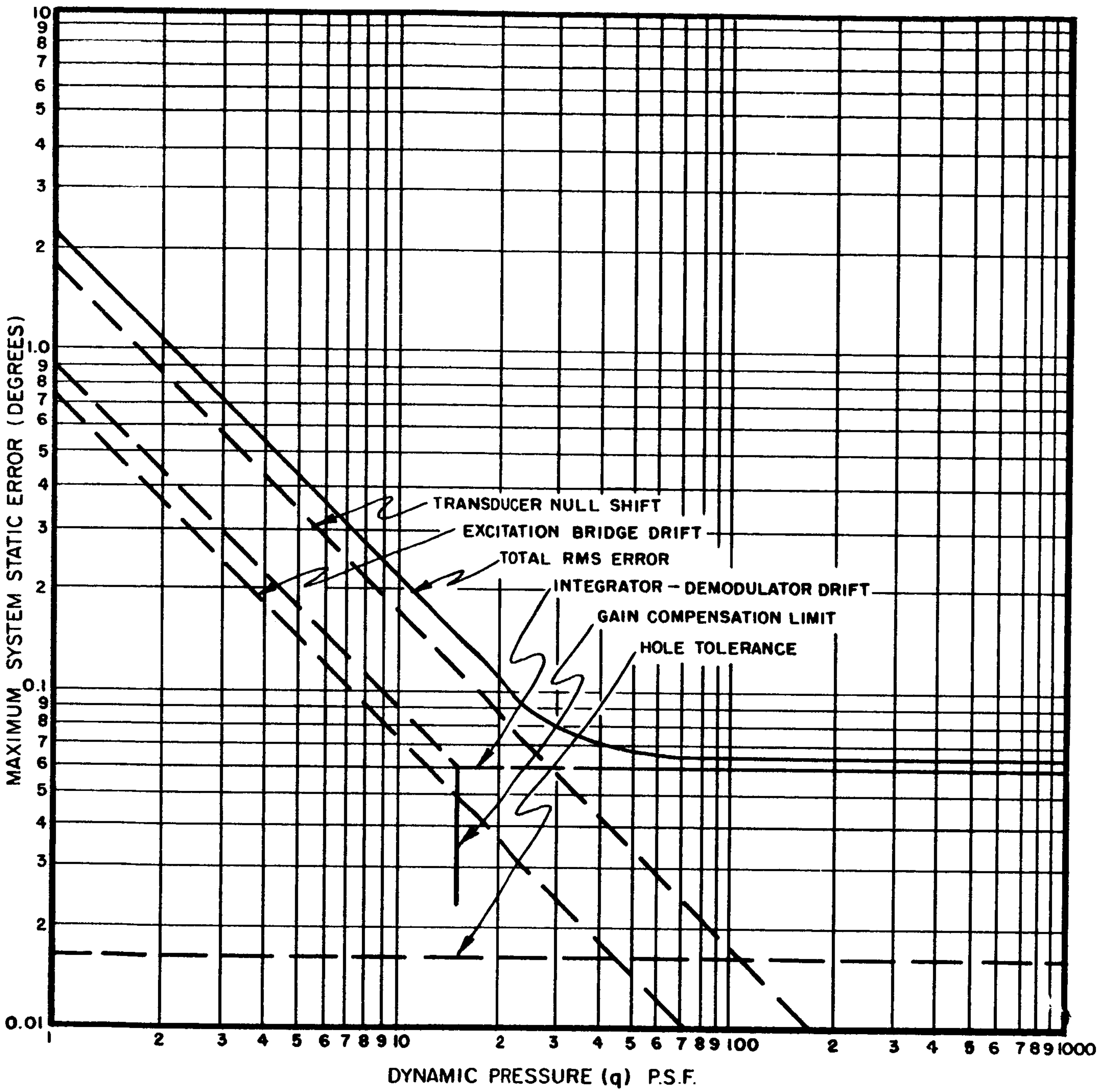

Figure 6-6. Ball Nose Static Accuracy

7. PRESENT COCKPIT DISPLAYS

Figure 7-1 shows the layout of the cockpit instruments used in a recent series of flights. Panel configuration changes for the testing of new indicators and indicator arrangements; therefore, little can be discussed about a standard instrument panel. However, the flight instruments shown in Figure 7-1 are typical of those presently in use and include the following:

a. Airspeed Mach indicator
b. Pressure altimeter
c. Angle-of-attack indicator
d. g meter
e. q meter (dynamic pressure)
f. Roll rate indicator
g. Inertial altitude indicator
h. Inertial speed indicator
i. Inertial rate-of-climb indicator
j. Attitude indicator (three-axis ball)

In addition to these indicators, newly developed displays are being tested in current and future flights. These displays include peak altitude prediction displays, ground track error indicators, on-board energy management displays, and moving tape indicators.

Physical and operating characteristics of the preceding indicators are summarized in Tables 7-1 and 7-2.

7.1 Attitude Indicator

Attitude information is presently displayed to the pilot by means of a three-axis all-attitude indicator similar to the one shown in Figure 7-2. This indicator contains a spherical ball

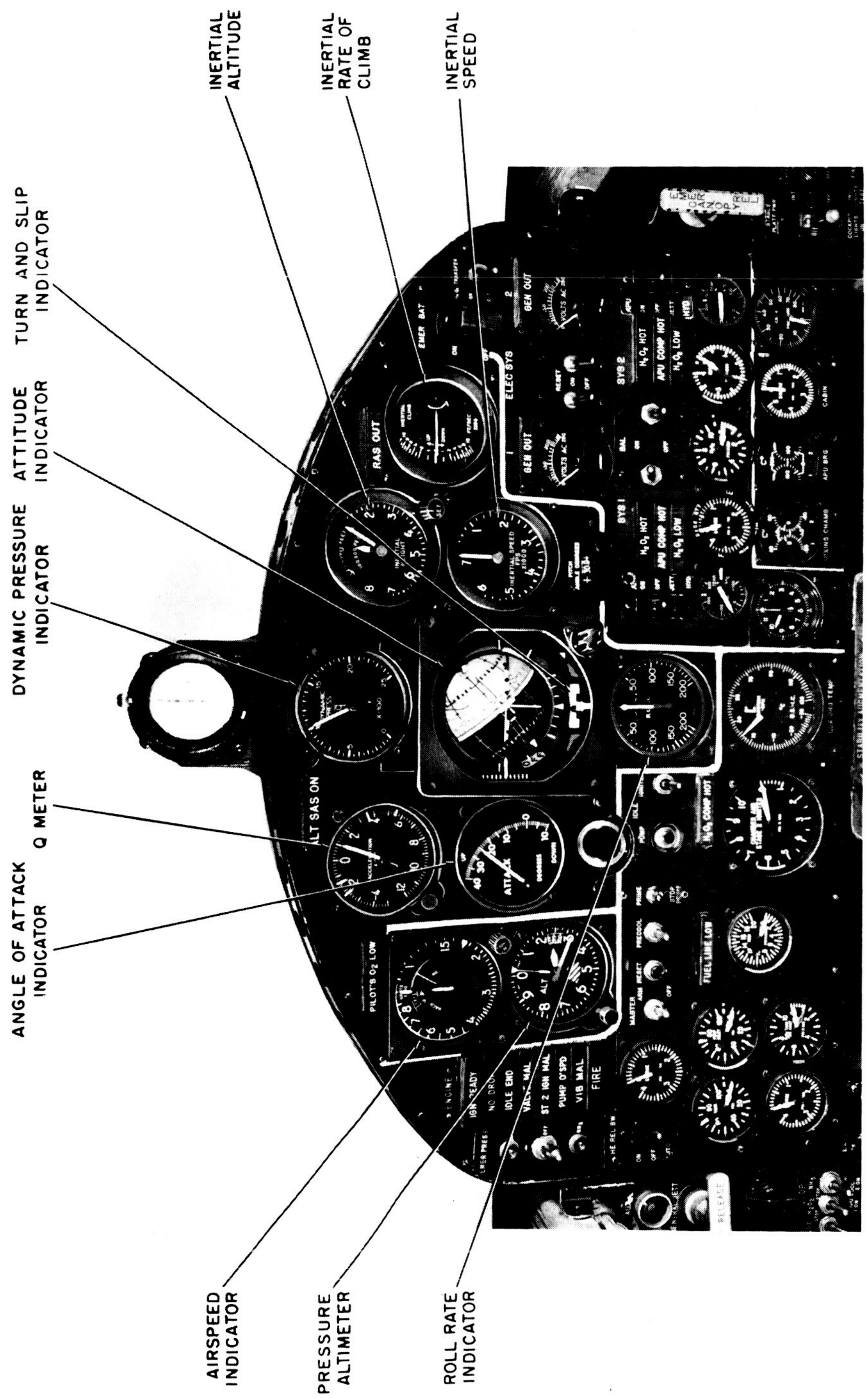

Figure 7-1. Instrument Panel Configuration

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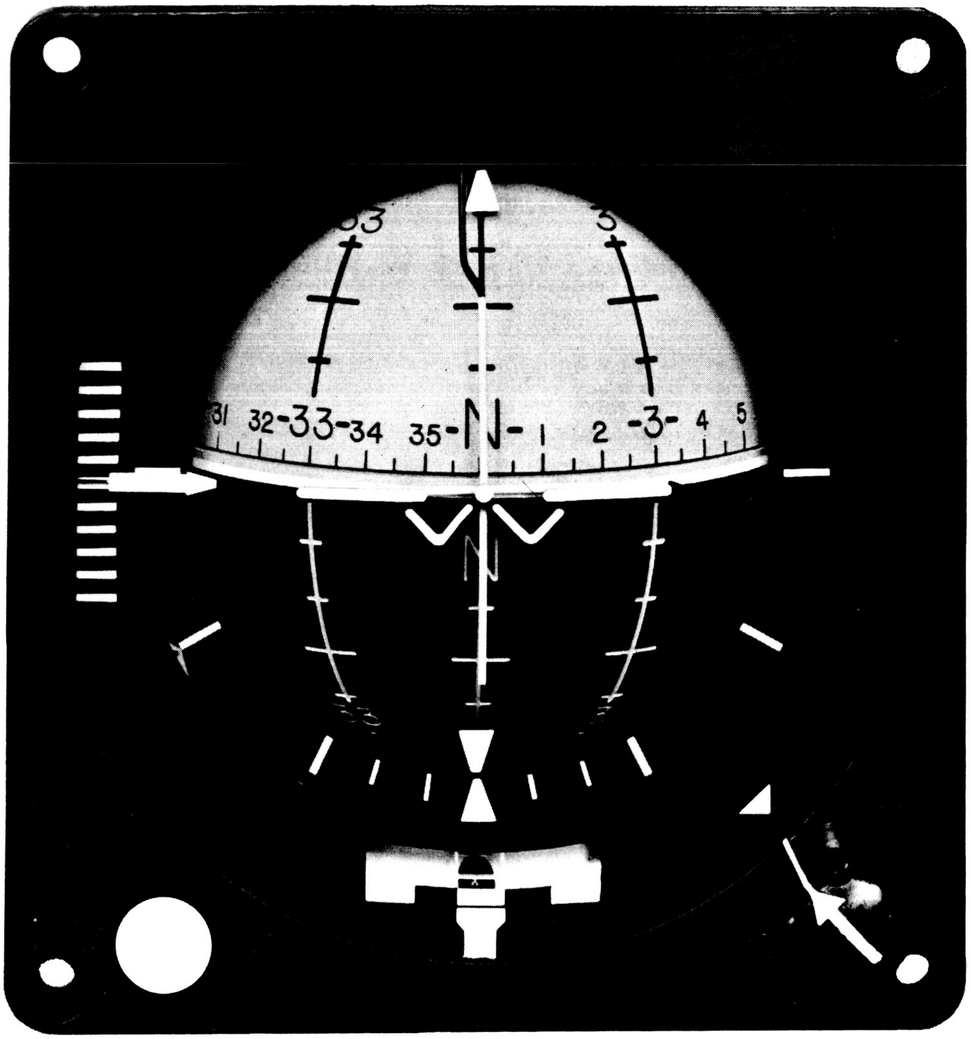

Figure 7-2. All-Attitude Indicator

approximately 3.5 inches in diameter with unlimited freedom in the three aircraft axes, and it is servo-driven from an on-board inertial platform. The sphere has a white horizon line with azimuth markings every 5 degrees. It is colored white above and black below the horizon. Vertical azimuth lines are included at every 30 degrees of azimuth with supplemental pitch markers every 10 degrees along their length. In a special modification for the X-15, additional azimuth markers have been provided on the ±30-degree pitch lines.

Superimposed vertical and horizontal crosspointers are used to indicate angle-of-attack and sideslip angle. A mode switching arrangement in the X-15 allows the vertical crosspointer to be used for displaying heading error instead of angle-of-sideslip. Typical sensitivities for these parameters are as follows:

a. Angle-of-attack: ±5 degrees full scale*
b. Angle-of-sideslip: ±12 degrees full scale
c. Heading error: ±12 degrees full scale

A vernier pitch indicator is located on the left side of the faceplate. It has a total range of ±5 degrees and displays pitch error from a command pitch attitude which is set in on a counter below the display.

Roll is displayed by a roll pointer which moves with the roll axis of the spherical ball. Readout is provided by reference markers on the faceplate at ±10, ±20, ±30, ±60, and ±90 degrees.

The operating characteristics of this indicator are summarized in Table 7-2.

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* Measured with respect to preset angle.

Table 7-2. Attitude Indicator Characteristics

	Performance
Parameters	Roll, pitch, and azimuth
Range	360 degrees in all axes, unlimited
Accuracy	± 0.5 degree at 0 degree, ± 1 degree otherwise
Follow-Up Rate	300 degrees per second roll and azimuth, 90 degrees per second pitch
Overshoot	3 degrees maximum
Scale Size	33 degrees per inch (0.03 inch per degree)
Minimum Division	5 degrees azimuth, 10 degrees pitch and roll
Marking Increment	10 degrees azimuth, 45 degrees pitch
Field-of-View	110 degrees pitch and azimuth
	Crosspointers
Parameters	Sideslip or heading (vertical), angle-of-attack error (horizontal)
Deflection	±1-1/16 inches
Response	1/3 second maximum
Linearity	7.5 percent of full scale deflection
Overshoot	1.5 percent maximum
	Displacement Pointer
Parameter	Pitch error
Range	±5 degrees
Accuracy	±0.5 degrees
Scale size	1-1/4 inches
Minimum Division	1 degree
Numbering	None
Response	2 seconds maximum
Linearity	7.5 percent of full scale

Table 7-2. Attitude Indicator Characteristics (cont.)

	Physical Characteristics
Panel Area	5 X 4.75 inches = 24 square inches
Display Area	4.3 inches diameter = 13.5 square inches
Depth	8.00 inches
Volume	190 cubic inches
Weight	9.5 pounds
	Environmental Conditions
Humidity	100 percent
Temperature	-54°C to +71°C
Pressure	30 inches Hg to 0.326 inch Hg (100,000 feet).
Acceleration	10 g maximum
Vibration	±1/2 degree max. error for double amplitudes of 0.020 to 0.100 inch
	Input Requirements
Electrical	Three 3-wire control transformer inputs for roll, pitch, and azimuth
Crosspointers	25 milliamperes into 1000 ohms full scale
Vernier Pointer	150 microamperes for full-scale deflection
Power	103 to 127 volts ac, 320 to 480 cps, not to exceed 35 VA.
	Reliability
MTBF*	1000 hours to 90 percent confidence level

* Mean-Time-Between-Failures

7.2 Angle-of-Attack Indicator (See Figure 7-3a)

The angle-of-attack indicator is a dial-type instrument located to the left of the attitude indicator (Figure 7-1). This instrument is often used as the primary indicator for control of the aircraft in aerodynamic flight. Angle-of-attack determines the lift-to-drag ratio of the aircraft and, therefore, is the prime means of controlling vertical velocity and, consequently, flight path angle.

The sensor for the angle-of-attack indicator is the hypersonic flow direction sensor (Figure 6-4). This device is extremely sensitive to airflow and, thus, will operate reliably at dynamic pressures down to 1 pound per square foot where the total sensor error approaches 3 degrees. Since the aircraft control surfaces require a minimum of 25 pounds per square foot for effectiveness, and the sensor error in this range is less than 0.1 degree, angle-of-attack is an accurate indicator of flow direction throughout the total aerodynamic portion of the flight profile.

This indicator has 50 divisions in 1-degree increments over the total range of 2.7 inches, giving a viewing sensitivity of 18.5 degrees per inch (Figure 7-3 a). A vernier indication of angle-of-attack, with a viewing sensitivity of 5.7 degrees per inch, is provided by the horizontal crosspointer of the attitude indicator.

7.3 Angle-of-Sideslip Indicator

The horizontal component of flow direction (sideslip) is presently displayed in the vertical crosspointer of the attitude indicator. The total range displayed is ±15 degrees using a deflection of ±0.875 inch. This gives a viewing sensitivity of 17 degrees per inch. Since there is no scale for use with this pointer, the pilot must estimate the actual sideslip angle in terms of the relative position of the pointer with respect to its full scale deflection.

Vehicle control has been adequate with this type of indication, since sideslip is usually maintained at zero in aerodynamic flight. A mode switch on the panel allows the inertial equivalent of sideslip (heading error) to be displayed on this pointer during ballistic flight.

This parameter has been shown to be extremely useful in the control and analysis of lateral stability under certain combinations of Mach number and angle-of-attack. This method of control is outlined in reference 6, Section 15. A discussion of the application of quickening techniques to the lateral control problem using sideslip angle is given in Appendix E, Paragraph 3.

Since the sensor for this parameter is also the hypersonic flow direction sensor, the accuracy of the display varies in the same manner as angle-of-attack.

7.4 Accelerometer (See Figure 7-3b)

The accelerometer, or g meter, is a dial instrument located above the angle-of-attack indicator (Figure 7-1). It indicates acceleration along the normal axis of the aircraft over a range of -5 to +12 g-units. This is displayed on a circular scale, 2.7 inches in diameter, corresponding to a viewing sensitivity of 2 g-units per inch.

This indicator is used to control the aircraft during pullup maneuvers (rotation about the lateral axis). The g-meter is the prime flight instrument for controlling vehicle stresses and heating rates during reentry, and for establishing and maintaining zero-g flight within the atmosphere. In the latter case, it is cross-referenced with the zero indication of the angle-of-attack indicator.

The sensor for the g meter is a self-contained acceleration-responsive element (mass) on a damping medium. The element is coupled mechanically to the pointer. No electrical output is provided by this instrument. Input signals for recording and telemetry of this parameter are provided by separate strain-gage type accelerometers.

7.5 Dynamic Pressure Indicator (See Figure 7-3c)

Indication of dynamic pressure (q) is provided by a dial instrument located directly above the attitude indicator. The instrument has a range of zero to 2500 pounds per square foot. The scale is divided into 500 increments of 50 pounds per square foot over a total circular range of 6.5 inches, providing a viewing sensitivity of about 400 pounds per square foot per inch. This instrument is accurate to within 3 percent at speeds above

Mach 2.5, with error increasing rapidly at lower speeds.

The q meter is used as the prime indication of vehicle heating rates. It also serves as a useful guide for flying at maximum lift-to-drag ratio when energy management is a prime consideration.

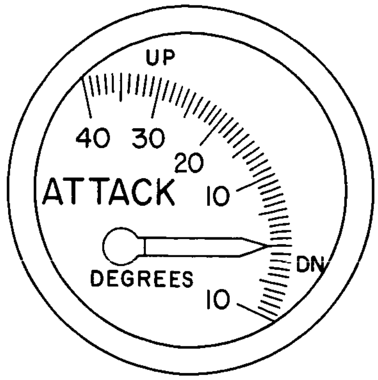
Pressure input for this indicator is provided by the ball nose (stagnation pressure port).

7.6 Inertial Indicators (See Figure 7-4)

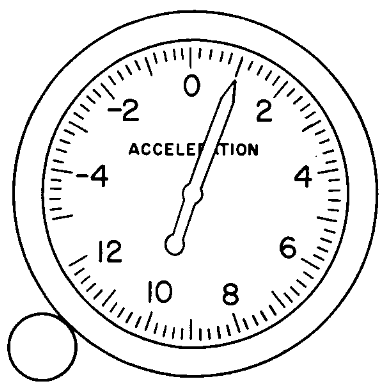
Display of three inertial qualities (height, speed, and rate-of-climb) is provided by the indicators shown in Figure 7-4. These instruments are located above and to the right of the attitude indicator as shown in Figure 7-1.

The inertial height indicator, (Figure 7-4a) is a two-pointer dial instrument with a maximum range of 1,000,000 feet. The scale is divided into 50 divisions, each representing 20,000 or 2000 feet. The indicator is driven electrically by the inertial platform computer, which performs the necessary integrations from vertical acceleration signals.

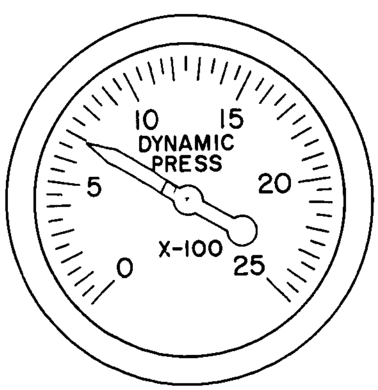
The accuracy of this indicator is limited only by the accuracy of the inertial flight data system and the reference height input provided before launch. Typical errors range from 5 percent at the end of 2.5 minutes to 5,000 to 10,000 feet after 10 minutes. Because of the double integration involved, height error increases exponentially with time. In spite of this effect, inertial height is used as a relative indication of flight status during a large part of the flight.



a.



b.



c.

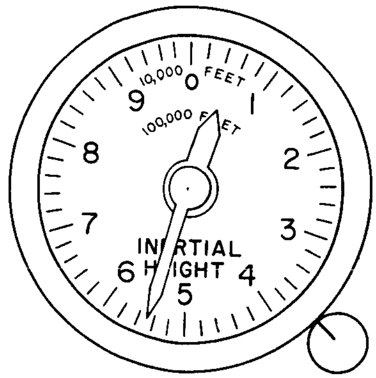
Figure 7-3. Flight Instruments(Partial)

The inertial speed indicator (Figure 7-4b) has a maximum range of 7000 feet per second. The scale contains 35 divisions giving an indication per division of 200 feet per second. This corresponds approximately to the accuracy of the velocity information obtained from the inertial system during the first 2.5 minutes of flight. This indicator receives velocity information from the platform in three components (North-South, East-West, and vertical) and contains circuitry for computing their vector sum for display of total velocity.

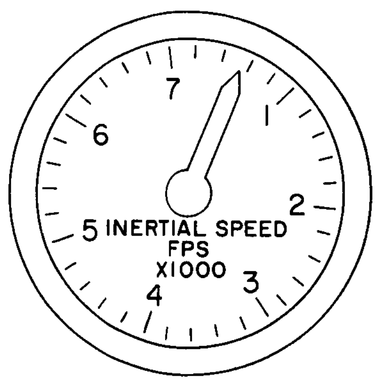
Inertial velocity is an important parameter for speed and heating flights where velocity must be maintained relatively constant during powered flight. In such cases, the pilot controls speed using the throttle and speed brakes to ±200 feet per second. In certain types of altitude flights, inertial speed is the prime cue for engine shutdown. Speed and height are cross-checked for an indication of status with respect to the planned profile during all types of flight.

Inertial climb (vertical velocity) is displayed on a dial indicator with a single horizontal pointer which is deflected upward for ascent and downward for descent (Figure 7-4c) The scale covers the range of 1000 feet per second in each direction with minimum divisions corresponding to 100 feet per second.

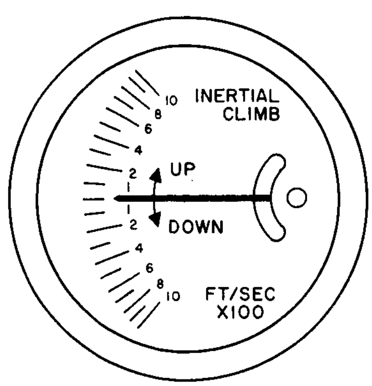
One of the problems with the indicator is that it does not have sufficient range to cover the total range of vertical velocities encountered (±4000 feet per second). Its primary use is in controlling rate of descent during reentry and in maintaining constant altitude for heating runs. In the latter case, deflections from null are used as a cue for changing angle-of-attack to correct deviations from level flight.



a.



b.



c.

Figure 7-4. Inertial Instruments

The inertial climb indicator remains sufficiently accurate around zero to be used throughout the entire flight.

7.7 Airspeed-Mach Indicator and Pressure Altimeter

The indicators for indicated airspeed (IAS) and pressure altitude will be discussed together since both are pressure instruments of standard military configuration. These instruments are located on the left side of the panel (Figure 7-1).

The airspeed-Mach meter has a single pointer and separate scales for airspeed and Mach number. The Mach scale rotates under the airspeed pointer plate and is visible through a cutout in the plate at airspeeds above 250 knots. The airspeed scale is nonlinear and covers the range of from 80 to 850 knots, with divisions every 10 knots up to 400 knots and every 50 knots thereafter. This suggests a readability ranging from 2.5 knots in the lower range to 12 knots in the higher range, based on a visual interpolation factor of 4 to 1. The accuracy of this indicator is a function of indicated value and ranges from ±2 knots to ±7 knots.

The pressure altimeter is a three-pointer instrument with a single scale ranging from zero to 10 divided into 50 divisions with numerals every 10 divisions. The pointers indicate increments of 10,000 feet, 1000 feet, and 100 feet, respectively. The most sensitive pointer, thus, indicates 20 feet per division.

The accuray of the pressure altimeter is a function of altitude and ranges from ±30 feet at zero altitude to ±280 feet at 50,000 feet. Above 50,000 feet, the error increases rapidly reaching ±1500 feet at 80,000 feet.

The pressure instruments are used only during subsonic periods of flight since indicator errors become appreciable above Mach 1 (Section 15, reference 5). This includes primarily the landing phase which begins at altitudes of 20,000 to 40,000 feet, and airspeeds averaging 300 knots.

Pitot pressure for the airspeed indicator and altimeter is supplied by the fuselage-mounted pitot head. Static pressure is supplied by ports on each side of the fuselage forward of the cockpit area.

7.8 Roll Rate Indicator

The roll rate indicator is located directly below the attitude indicator (Figure 7-1). It has a range of ±200 degrees per second with minimum divisions of 10 degrees per second. The indicator is normally accurate to within 2 to 3 degrees per second throughout the entire flight. It is driven by a self-contained electrically operated gyro.

The roll rate indicator is used primarily to maintain a given roll attitude (usually zero degrees). It has also been used to control roll and yaw oscillations under conditions of simulated damper failure.

7.9 Timer

Timing cues for the various portions of flight are provided by a stopwatch mounted directly above the q meter (Figure 7-1). The second-hand has a range of 100 seconds per revolution. A smaller hand counts 100 second increments from zero to 30, giving a total range of 3000 seconds. The stopwatch is started by an electrical signal at the time of main propellant valve opening, thus measuring time from launch.

Absolute time is displayed by an eight-day clock located in the lower portion of the instrument panel with the engine instruments (Figure 7-1).

8. INTEGRATED DISPLAY DESCRIPTION

8.1 Criteria for Design

The contact analog method of display integration was selected to serve as the basis for the development of an advanced integrated display system. A specific design format evolved from the detailed analysis of the mission profile and pilot task requirements presented in Sections 3 and 9. This design was based on human-factors considerations derived from literature of display technology. This literature was obtained from human-factors engineers, from the experience of Norden engineers in the laboratory, and perhaps most significant, from the direct experience of users of existing contact analog equipment who have evaluated similar designs in various types of vehicles and conducted controlled experiment to evaluate the effectiveness of specific display elements.

Specific design criteria used in the development of the integrated display format include the following:

a. Consistent manner of indication for the several kinds of data
b. Selection of easily-discriminated symbology
c. Legibility from normal head-position of the pilot
d. Reduction in the number of displayed data categories
e. Standardization of location and the avoidance of confusion among scales and their pointers
f. Qualitative representation of parameters where accuracy of interpretation is not necessary
g. Reference scales for those parameters where precision is required
h. Inclusion of significant visual cues in analog form

8.2 Display Elements and Format

8.2.1 Generation I Display Format

The following is a discussion of important flight parameters that will be presented on the "first generation" integrated display. The display format is oriented toward the control of an X-15 type aircraft flying a typical profile that includes climbout, peak altitude reentry and landing phases.

Aspect Ratio and Reticle (See Figure 8-1)

A 4 to 3 (vertical/horizontal) aspect ratio was selected for the CRT display, since this permits a large viewing angle in the vertical plane (55 degrees). This viewing angle is required, since there is a greater need for display information in the vertical rather than the horizontal plane. The horizontal viewing angle is 40 degrees.* An electronically generated reticle with an open center, which represents the fore-aft axis of the vehicle, is positioned slightly above the center of the screen. The combined effect of the large vertical viewing angle and the offset reticle tends to keep the horizon line and angle-of-attack symbol on the viewing screen throughout the flight.

Pitch Angle and Roll Angle (See Figure 8-1)

The horizon line pitches and rolls about the reticle in a

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* For purposes of comparison, the calculated field-of-view for the X-15 aircraft is approximately 180 degrees horizontal with a 17 degree dead zone directly ahead of the pilot, and ranges from 17 to 26 degrees vertically.

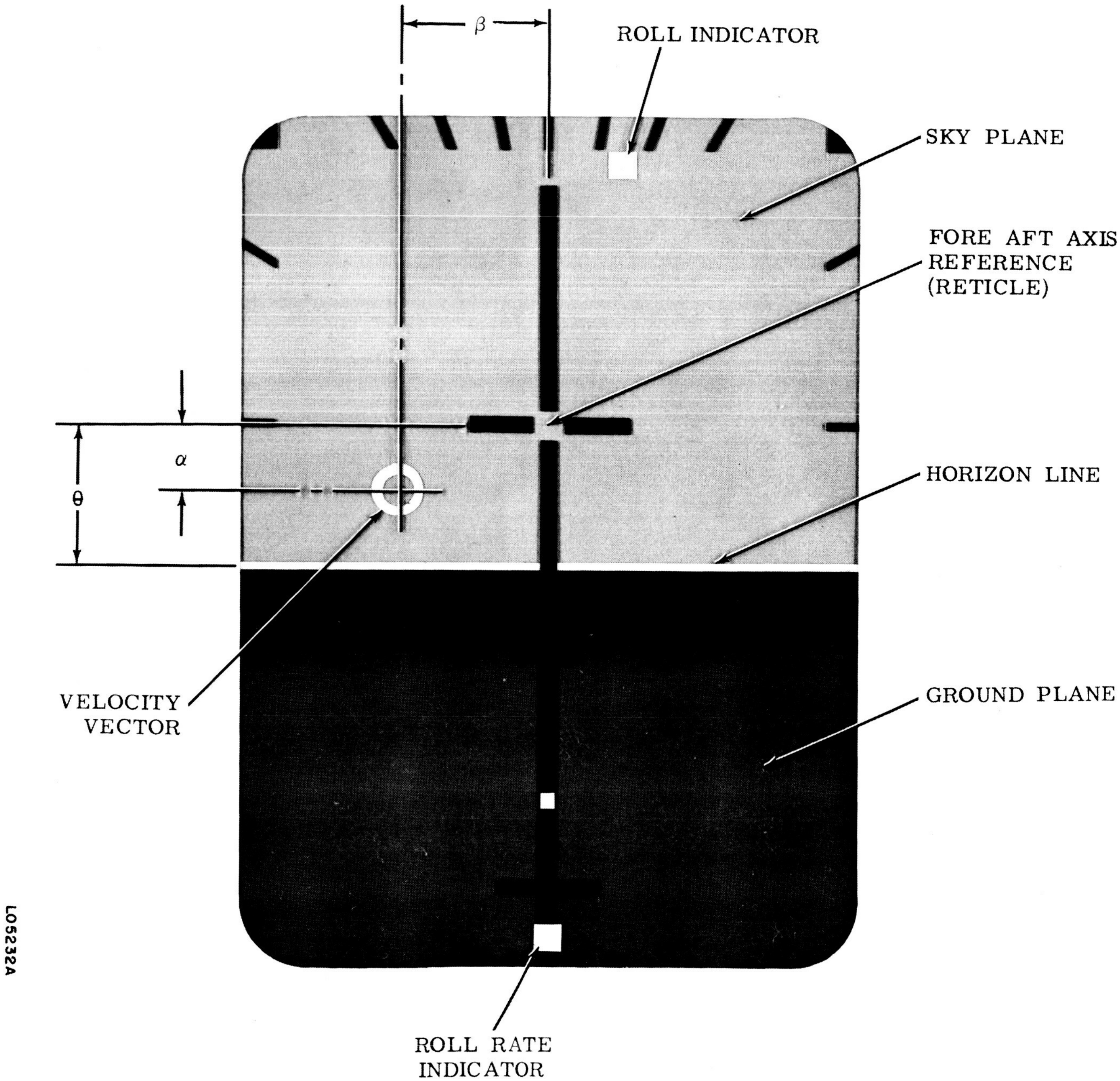

Figure 8-1. Integrated Display Elements I

mathematically correct manner through a full 360-degree pitch angle (θ) or roll angle (ϕ). The shading of the sky plane (area above horizon line) and the ground plane (area below horizon line) give a real-world effect to the display. If a more elaborate ground texture is desired, a grid, quasi-random, or checkerboard type ground texture can be provided. These types of ground textures provide altitude, heading, and ground velocity cues in true perspective. For example, the squares of the checkerboard pattern are programmed to become smaller as altitude is increased, and become larger as altitude is decreased. In addition, they translate with North-South and East-West velocity inputs. See Figure 9-7 for an example of a textured ground plane.

Ground texture has not been included in the recommended X-15 display format, since relative heading and altitude information are provided by the command heading cursor and pathway, respectively. Human factors considerations pertaining to ground texture indicate relatively poor accuracy for altitude cues furnished by this means. In the landing phase, where accurate altitude cues are imperative, ground texture could possibly be provided by a forward-looking TV camera.

Additional pitch information is provided by pitch reference markers (see Figure 8-2) located at +30, -30, -60, and +60-degree pitch angles. These markers pitch and roll about the reticle as does the horizon line and are therefore coded so that they cannot be mistaken for the horizon line during extreme pitch attitudes. A pitch cursor is also provided (see Figure 8-2). This cursor is programmed at a critical pitch angle (usually the climbout angle). The pilot is required to keep the pitch cursor on the reticle during climbout.

For quantitative indication of roll angle, a separate

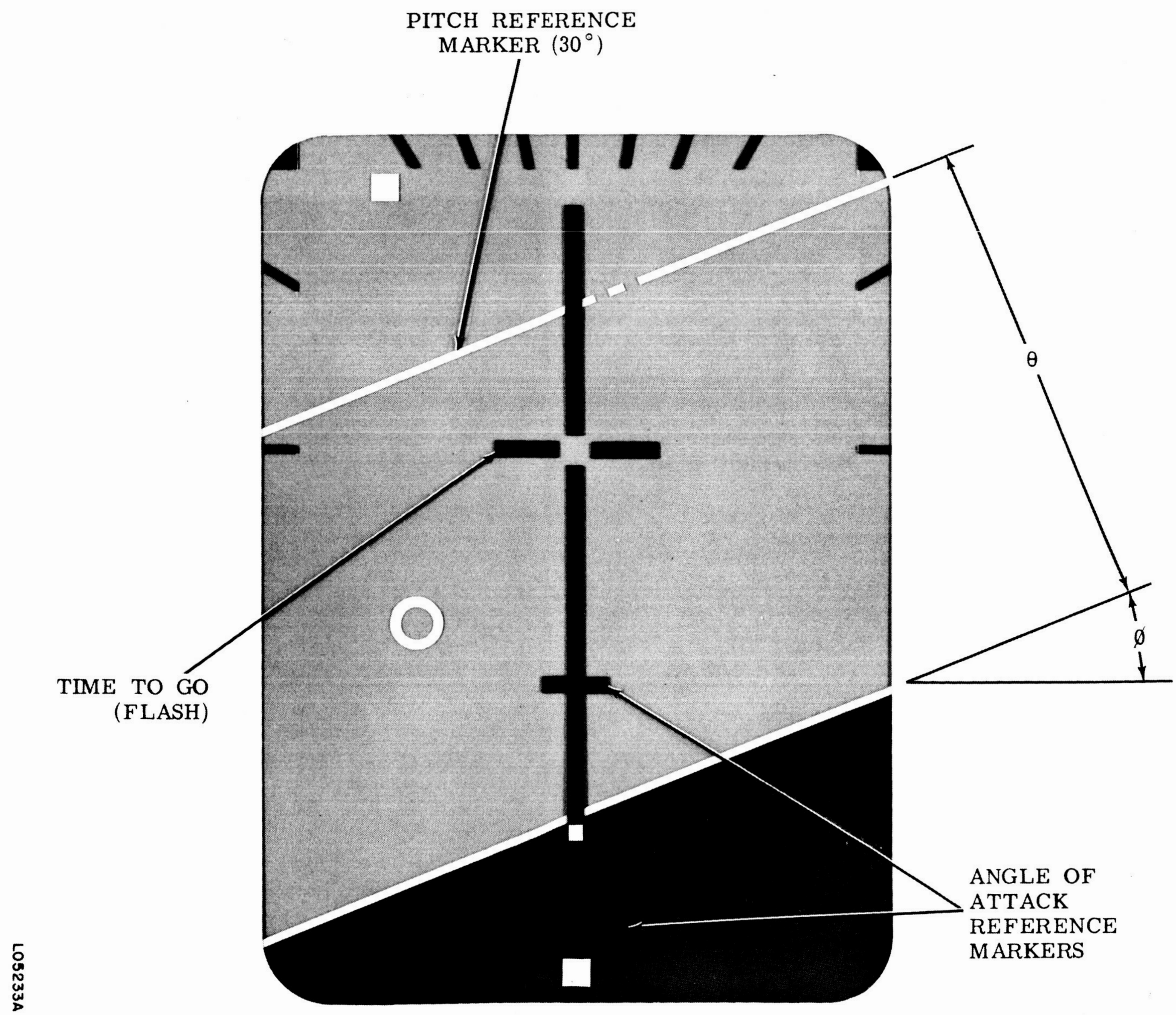

Figure 8-2. Integrated Display Elements II.

pointer is provided in screen coordinates with freedom only in roll. Reference markers are included at 0°, $\pm 10^\circ$, $\pm 20^\circ$, $\pm 30^\circ$, $\pm 45^\circ$, $\pm 60^\circ$, and $\pm 90^\circ$. The 0° marker is electronically generated for improved accuracy, while the remaining markers are in the form of an overlay. A similar pointer and scale is provided at the bottom-center of the screen for indication of roll-rate (Figure 8-1).

Angle-of-Attack and Sideslip Angle

Angle-of-attack (α) and sideslip angle (β) position the velocity vector symbol vertically and horizontally with respect to the reticle symbol. This vector is represented on the viewing screen by a small circle (see Figure 8-1). Angle-of-attack has a range of -20 to +35 degrees and sideslip has a range of ± 20 degrees. The pilot determines these angles by relating the position of the velocity vector to the reticle. Angle-of-attack reference markers are located on the vertical reticle line at 15° and 30° (see Figure 8-2).

Flight-Path Angle

As a consequence of displaying angle-of-attack in the mathematically correct screen position, flight-path angle (γ) can be read-out as the distance between the velocity-vector symbol and the horizon line. An input for this parameter is not required when angle-of-attack is available (paragraph 8.3).

Heading Deviation

Heading deviation from a preset heading is indicated by a heading cursor (Figure 8-3). The cursor has freedom in pitch, roll, and heading. The pilot would be required to keep the cursor close-to or on the reticle center. Additional heading reference markers

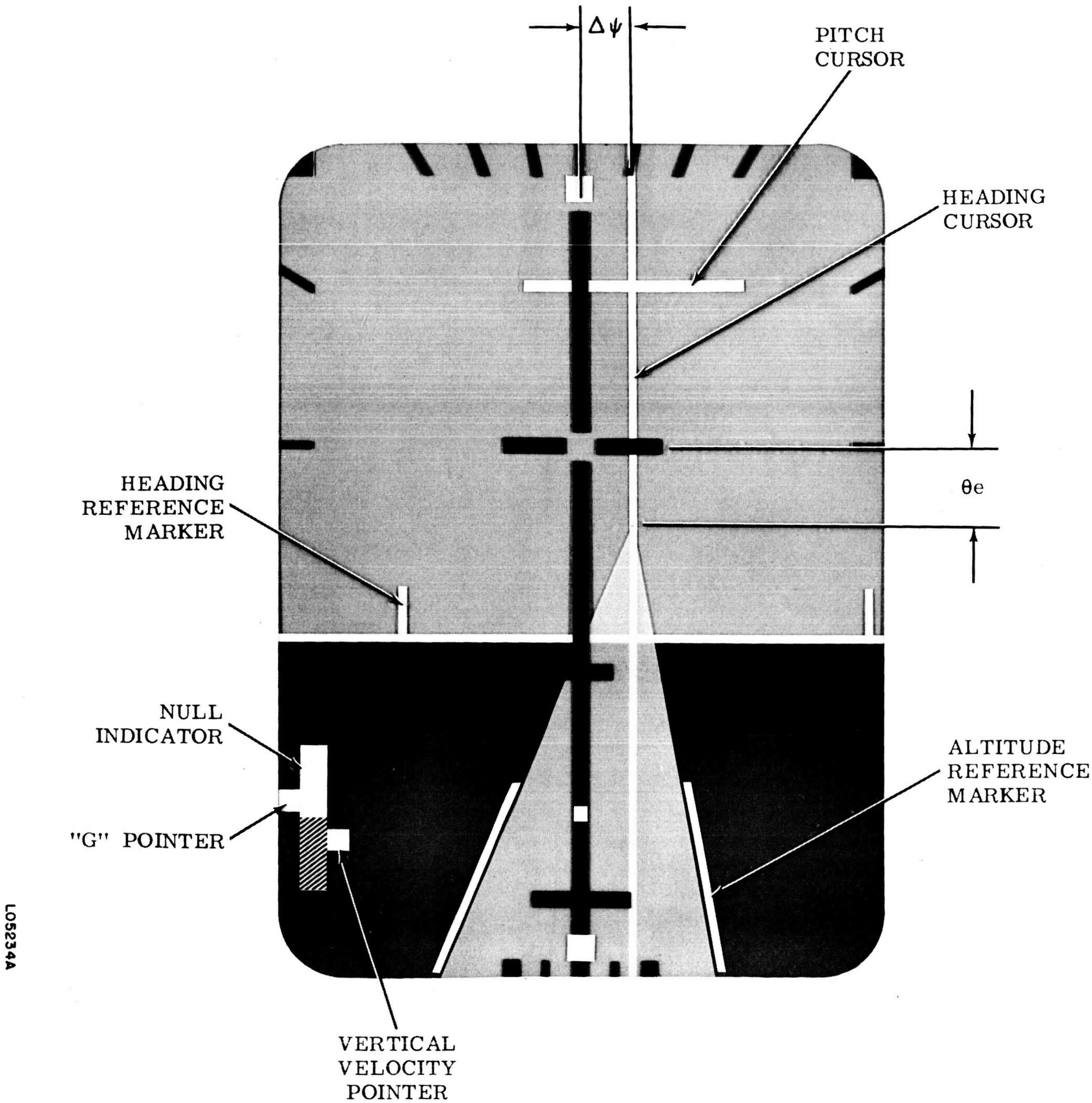

Figure 8-3 Integrated Display Elements III

are located on the horizon line at +15 and -15 degrees away from the heading cursor (in that order) to supply a reference for large deviations in heading.

Acceleration and Vertical Velocity

An electronically generated vernier is provided on the left side of the display (Figure 8-3). The white pointer on the left of the vernier gives a null indication when a preprogrammed value of g is attained by the vehicle. Similarly, the white pointer on the right of the vernier nulls when a preprogrammed value of vertical velocity (V_v) is attained. Hence, the pilot has a null readout of g and V_v at times during the flight profile when these parameters should be held to close tolerances.

Since several values of g and V_v may be required during a flight, either manually switched or preprogrammed mode switching signals will be provided to select the proper null value of g and V_v.

Phase Warning

For critical timing functions (i.e., time to burn out), the reticle will be programmed to flash in a coded manner 4 seconds prior to the change in phase.

8.2.2 Generation II Display Format

In the "second generation" display (Figure 8-3) a command pathway is added to indicate trajectory errors. The pathway tip is positioned in the reticle center for zero error. Vertical displacement of the pathway tip is an indication of pitch-error. Horizontal displacement of the pathway tip indicates heading error.

Altitude error, which is defined as the difference between the actual vehicle altitude and the commanded or programmed altitude, is displayed as the deviation between two altitude reference markers and the edges of the pathway. For zero altitude error, the altitude reference markers are positioned along the edge of the pathway; for positive altitude error, the pathway becomes narrow and the reference markers appear outside the pathway; for negative altitude error, the pathway becomes wider and the reference markers appear within the pathway. For large negative values of h_e, the pathway will completely fill the display screen. The pathway is, therefore, limited at a predetermined value of h_e.

<u>Other Pathway Programs</u>

An alternate approach to presenting command information with the pathway would be to program the path tip with commanded angle-of-attack and command sideslip angle. To fly this pathway, the pilot would be required to keep the pathway tip inside the small white circle representing the actual angle-of-attack and sideslip angle.

If the requirement exists, the near-end of the command pathway can be programmed with cross-track error. The near-end of the pathway would move toward the right corner of the display when the vehicle was to the left of the command course or to the left corner of the display if the vehicle were to the right of the command course.

If velocity cues are desired, tarstrips appearing in true perspective, can be added to the pathway. These tarstrips move toward the pilot in proportion to total vehicle velocity. Although truly quantitative velocity information is not available with this means of display, reasonable estimates of velocity can be obtained.

8.3 Input and Computer Requirements

A listing of the input requirements for the display generator appears in Table 8-1. These inputs must appear as d-c voltage analogs of the parameters received by the display generator. Signal converters are, therefore, necessary to convert the sensor and computer outputs into the proper form. As an example of signal conversion, consider the requirement for Sine Θ and Cosine Θ. A synchro on the inertial platform transmits a signal proportional to Θ. The signal converter must take this signal and drive a transolver to develop 400 cycle a-c signals whose peak amplitudes are proportional to Sine Θ and Cosine Θ. These signals are demodulated and filtered to obtain analog d-c signals proportional to Sine Θ and Cosine Θ. The command pathway is positioned on the display screen by altitude error (h_e), pitch error (Θ_e), and heading deviation ($\Delta\Psi_h$). A computer must be available to solve the following equations:

$$h_e = h - h_c \tag{1}$$

$$\Theta_e = \Theta - \Theta_c \tag{2}$$

To solve this, the computer must compare the actual flight parameters with a preprogrammed flight profile (commanded parameters), or in the case of heading and climbout pitch, a constant angle. Climbout pitch is set-in by the pilot on the PITCH ANGLE SET CONTROL on the panel. The proper heading reference angle for each flight must be set-in in a similar manner, either by the pilot or ground crew. Means must then be provided for obtaining the difference of the two synchro angles.

The presently used $\Delta\Psi$ mode employs a similar technique for obtaining a heading deviation reference. In this case, an auxiliary synchro (slaved to the stable platform heading synchro)

Table 8-1. Integrated Display Input Functions

FUNCTION	DESIGNATION	ELEMENTS CONTROLLED	SENSOR
Vehicle Heading Deviation	Sin Cos $\Delta\Psi_h$	Heading cursor, pathway tip	Platform
Vehicle Pitch	Sin θ Cos θ	Pitch of horizon line, ground and sky plane	Platform
Vehicle Roll	Sin ϕ Cos ϕ	Roll of horizon line, ground and sky plane, heading cursor, pathway	Platform
Vehicle Angle-of-Attack	Sin α Cos α	Vertical position of velocity vector	Ball nose
Vehicle sideslip Angle	Sin β Cos β	Horizontal position of velocity vector	Ball nose
Vehicle Altitude Error	h_e	Pathway width	Computed
Vehicle Pitch Angle Error	Sin θ_e	Vertical position of pathway tip	Computed
Climbout Pitch Angle	Sin θ_c Cos θ_c	Climbout pitch cursor	Panel control
Acceleration	g	Acceleration Vernier	Accelerometer
Vehicle Vertical Velocity	V_v	Vertical Velocity Vernier	Platform
Display Mode Switch	k_s	Vernier Pitch cursor	Computed or manual
Phase Warning Pulse	P.W.	Reticle cross-bar flashing	Computed

is disconnected when the mode is entered. The difference-angle between the platform heading-synchro and this reference synchro is displayed as $\Delta\Psi$.

The input signal for the g pointer is derived directly from the electrical output of an accelerometer of conventional design. The proper null levels are preset electrically with potentiometers before comparison takes place in the analog circuitry. When more than one null value is desired, a suitable means of switching reference voltages must be provided, controlled either by the pilot or on the basis of some preselected parameter (such as time from launch).

The input for the vertical velocity pointer is obtained directly from the inertial flight-data system in the form of a potentiometer resistance (see Appendix B). If a null value other than zero is desired, a reference voltage is inserted before comparison. The reference could also be programmed in terms of a selected parameter, or by the pilot or ground crew.

Signals which are analogs to α and β are available from the ball nose sensor. Above the atmosphere however, the ball nose is inoperative and these outputs are not present. We can, however, define these angles in space as:

$$\alpha' = \theta - \gamma \tag{3}$$

$$\beta' = \psi_h - \psi_c = \Delta\psi_h \tag{4}$$

where θ and $\Delta\Psi_4$ are available directly and can be determined by solving an equation of the following type:

$$\gamma = \sin^{-1} \frac{V_v}{\sqrt{V_v^2 + V_{NS}^2 + V_{EW}^2}} = \sin^{-1} \frac{V_v}{V} \tag{5}$$

α' and β', defined by equations (4) and (5) can be used to position the velocity vector when the vehicle is out of the atmosphere. This indication of velocity vector in inertial space might allow the pilot to set up the critical reentry parameters (angle-of-attack and sideslip) earlier in the reentry phase, rather than having to wait until the ball nose sensor becomes active.

Equations (1) through (5) require a flight computer which is not presently aboard the X-15 vehicle. However, the present on-board computing equipment is sufficient to display all but the command pathway and inertial velocity vector.

The selection of a digital flight computer would require a detailed analysis of an optimum interface for the integrated display. The display presents both actual (status) and command information to the pilot. Control studies indicate that operator performance is seriously degraded by the introduction of any time delay in the presentation of status information. If sensor data is processed by the central computer, this data should be updated with respect to the prime sensor, to adequately reflect status information in the display. Command information presented in the display equipment is composed of error indications which are the differences of command values and status values. The status values, which are related to the vehicle attitude fluctuations, are rapid and should be displayed without delay. The command values normally change slowly and can therefore tolerate delays that may be introduced by the cycling rate of the digital computer.

A second consideration in the selection of an optimum interface for the display equipment is that of emergency or alternate mode operation. As presently designed, the integrated display is capable of displaying all status information directly from primary sensors (or signal converters). In the event of other avionic failure, the pilot will continue to have a maneuvering display in the same format and on the same indicator as under normal conditions.

A diagram of input requirements is shown in Figure 8-4.

9. ALTITUDE MISSION ANALYSIS

9.1 Profile Definition

A recent altitude flight profile has been selected to serve as the basis for a detailed analysis of pilot control tasks. The flight plan for this mission is given in Appendix C. The profile includes a peak altitude of 360,000 feet, a maximum velocity of 5400 feet per second, a reentry roundout acceleration level of 5.2 g, maximum dynamic pressure of 1200 pounds per square foot, and ground track distance of 325 miles.

The important parameters of this flight are shown graphically in Figure 9-1.

This flight can be divided into five phases, each with different characteristics and control requirements. A description of each phase is given in Table 9-1.

9.2 Pilot Task Analysis

Information from the flight plan has been correlated with data gathered from pilots and operations personnel in the development of the Pilot Task Analysis Chart shown in Figure 9-2. The application of this method of analysis to several mission profiles proved to be a useful aid in the development of the first and second generation integrated displays.

9.3 Flight Description

The altitude flight will now be described in terms of the Generation II integrated display outlined in Section 8. For task-parameter relationships, see Figure 9-2.

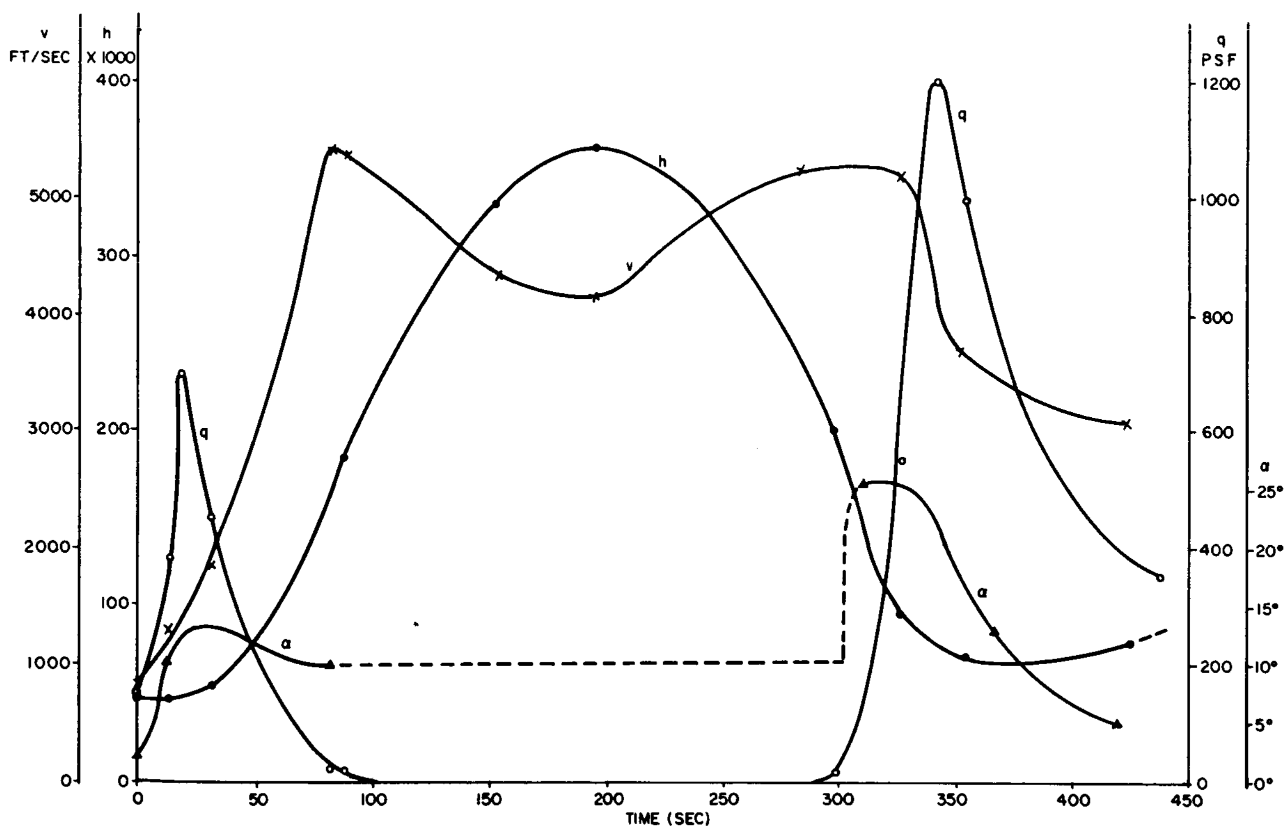

Figure 9-1. Altitude Mission Profile

Table 9-1. Flight Phase Requirements and Characteristics

PHASE	DESCRIPTION
1. Launch	Begins with release from B-52 Terminates with engine ignition
2. Boost	Begins with engine ignition (clock begins)
Rotation	Terminates upon reaching climb out pitch angle
Climb	Begins with climb out pitch angle; terminates with loss of thrust
3. Ballistic	Begins when dynamic pressure falls below 10 pounds per square foot; includes peak altitude; terminates at dynamic pressure of 10 pounds per square foot
4. Reentry	Begins at onset of sensible atmosphere $q = 10$ pounds per square foot $h \cong 220{,}000$ feet Includes maximum pressure; terminates when rate of descent falls below specified value
5. Terminal	Begins at termination of reentry roundout; terminates at completion of landing

Prior to launch, the pilot must prepare the engine and check all instrument systems, as well as performing other checks. (For simplicity, only those tasks which depend primarily on the display parameters under consideration will be described). Part of the instrument check includes an operational test of the ball-nose sensor. A test button is depressed, causing the ballnose to deflect to a predetermined number of degrees in each axis. The pilot checks the corresponding deflection of the velocity vector symbol.

At approximately two minutes before launch, the engine is precooled and primed. At thirty seconds before launch (or less), the pilot begins the igniter idle phase in which the igniter plugs are energized. The engine is then ready for launch.

The drop from the mother ship is initiated by the X-15 pilot, either at his own descretion or by a countdown from the B-52 pilot. Upon release, the throttle is pushed to 100-percent thrust causing the engine to fire. Time-from-launch is measured with respect to the opening of the main propellant valves by a timer (paragraph 7.9).

Immediately after launch, the pilot is concerned with assuring himself that the engine is performing properly. Primary cues for engine operation are provided by the chamber and fuel pressure indication. Once proper operation has been established, the pilot's attention shifts to controlling the aircraft attitude in accordance with the established rotation procedure. This is accomplished by establishing and holding a 10 to 12 degree angle-of-attack until the desired two-g normal-acceleration is attained. Referring to Figure 9-3, the velocity vector symbol (circle) is maintained slightly above the 15-degree reference line until the left-hand pointer begins a downward movement toward null (two g's).

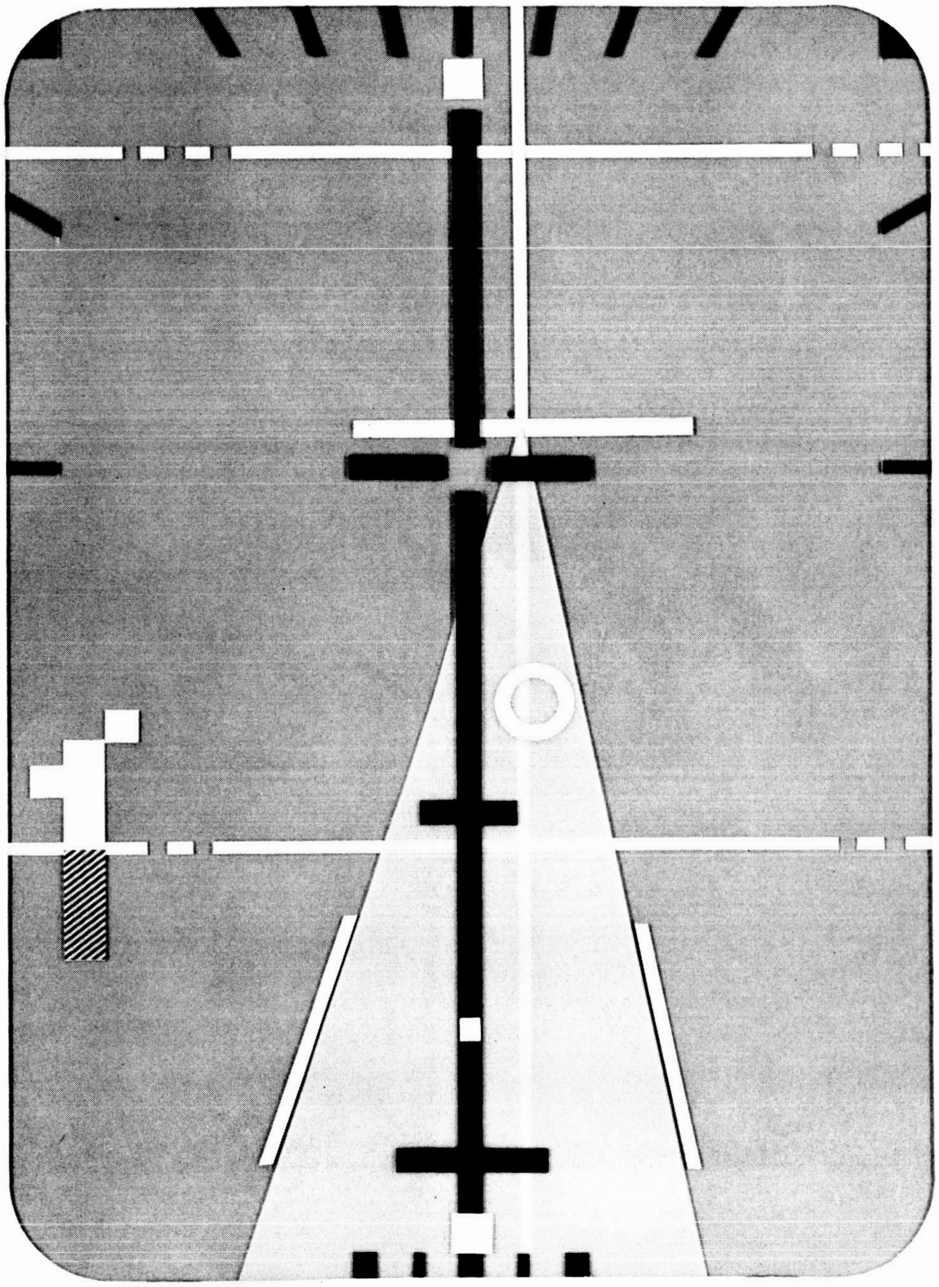

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Figure 9-3. Constant Pitch Climbout.

When the pointer reaches the null indication, the pilot's attention shifts to maintaining the two-g null while holding his heading constant. Roll and sideslip indications are used to minimize heading changes during climbout. The path tip at this time is programmed with the climbout pitch angle and, therefore, is off the top of the screen (since the path is transparent, no information is lost).

Visual outside reference is lost when the horizon disappears from the cockpit field-of-view.

While maintaining two-g null, the pilot observes the path tip (and climbout pitch-reference line) as it moves down the screen. In the general case, the path tip and pitch reference line would be programmed separately.

It should be noted that no vernier pitch indication is required on this display, because the normal display sensitivity (8.6 degrees per inch) is sufficient.

During the climbout phase, vertical trajectory errors are displayed by changes in the width of the command path with respect to the altitude reference markers. This supplements error information received from the ground controller by radio link. The pilot continuously monitors the special panel indicator for predicted (peak) altitude as an additional cue for trajectory correction.

Engine shutdown is extremely critical to the successful accomplishment of an altitude mission. In the profile under consideration (Figure 9-1), the engine was programmed to burn for 84.5 seconds for a peak altitude of 360,000 feet. At a predetermined time interval before burnout, the pilot will receive a warning of engine shutdown by periodic flashing of the reticle. During this time, he must decide whether or not to manually shut down the engine, based on the peak altitude prediction indicated.

After shutdown, the pilot has no further control over the vehicle trajectory, since the dynamic pressure at this time is no longer sufficient for control surface effectiveness. Vehicle attitude, however, can be controlled using the reaction jets in the nose and wings.

Just before leaving the atmosphere, the pilot establishes zero sideslip and switches to $\Delta\Psi$ mode. The horizontal displacement of the velocity vector symbol is then proportional to the difference between present vehicle heading and the heading before the mode was entered (refer to paragraph 8.3). The pilot then has a reference for re-establishing zero sideslip for reentry.

Approximately 36 seconds before peak-altitude occurs, the pilot positions the vehicle to a level attitude and rolls to a 45-degree bank angle. The display corresponding to this situation is shown in Figure 9-4. Note that the path tip commands the zero pitch angle and the velocity vector symbol is in the center of the reticle, corresponding to $\alpha = 0$, $\Delta\Psi = 0$, and $\gamma = 0$. After the vehicle reaches peak altitude, altitude error information is no longer displayed on the path.

At 220,000 feet, the pilot must establish the proper reentry angle-of-attack (26 degrees) using reaction jets in the nose of the vehicle. Figure 9-5 indicates the display picture at the time of this reentry attitude maneuver. For generality, a small heading error is shown. A reference mark has been included to indicate the command reentry angle-of-attack. This mark is a square with one-degree sides. The circle representing velocity vector has a two-degree inner diameter, thus enabling the proper angle-of-attack

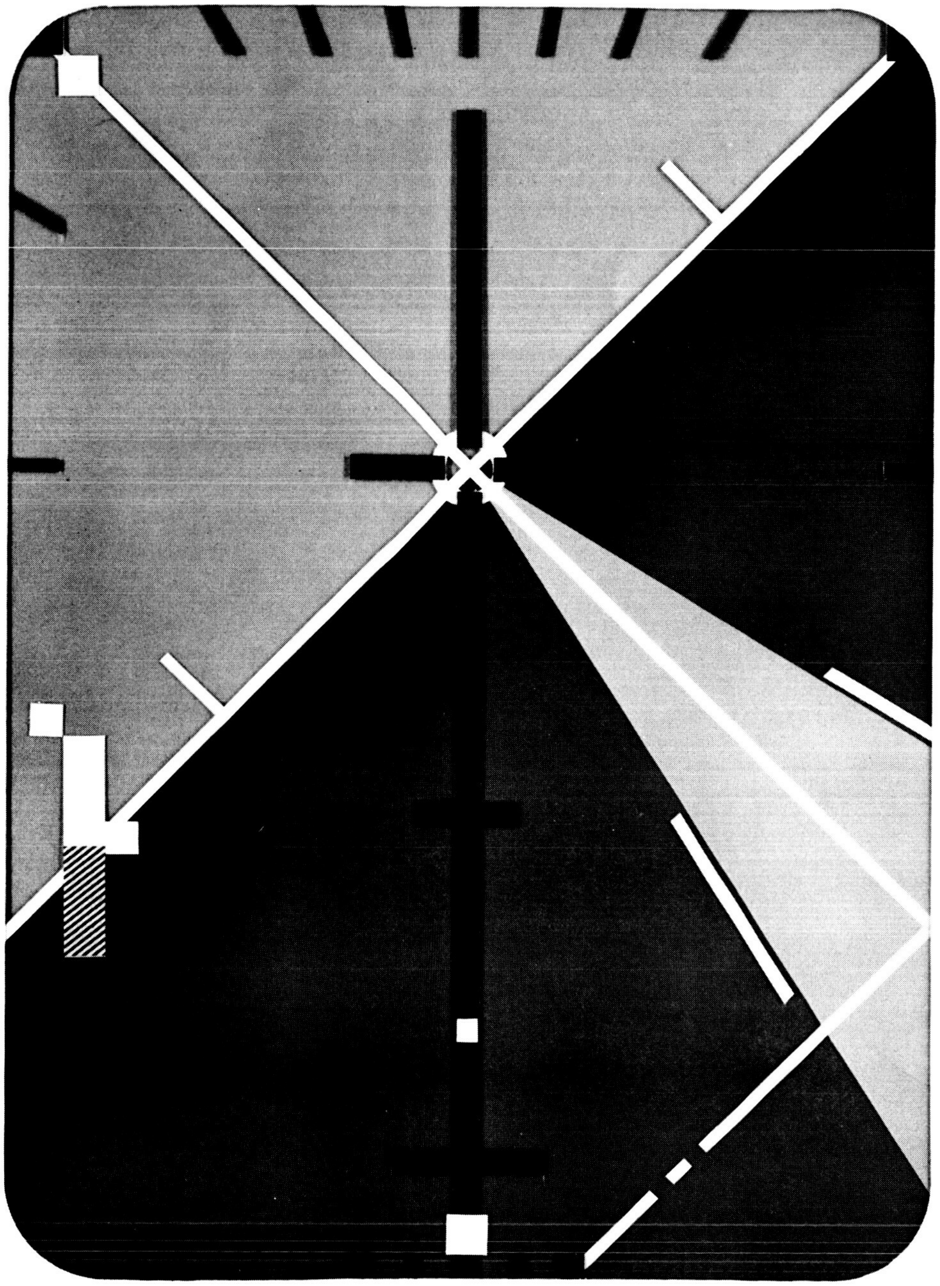

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Figure 9-4. Peak Altitude

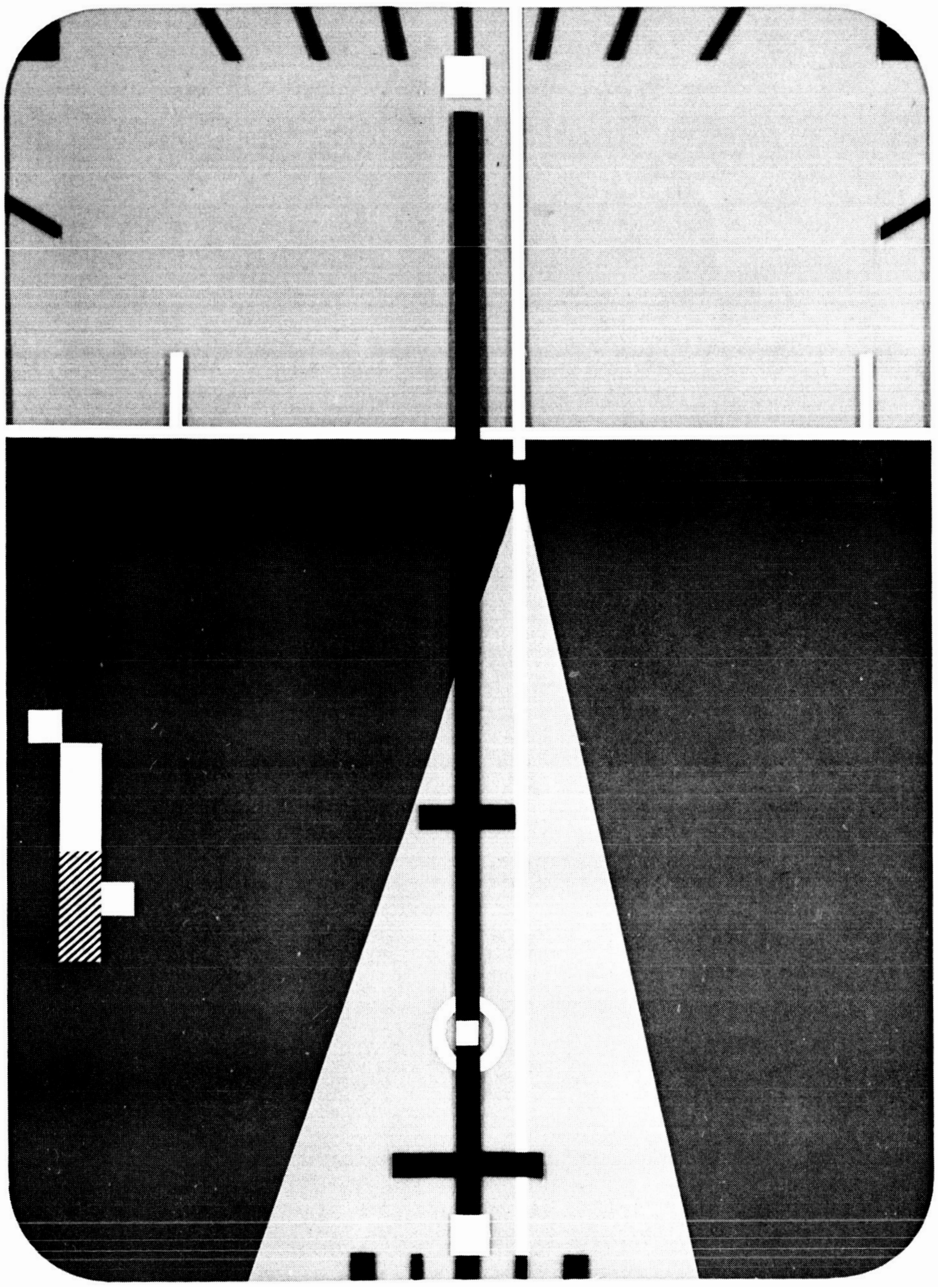

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Figure 9-5. Establishment of Reentry. Attitude.

to be established within better than one degree. As a result of the high reentry angle, the vehicle is pitched only slightly downward with respect to the horizon.

After the aircraft passes an altitude of 200,000 feet, the velocity vector horizontal input is switched back to the STANDBY mode, for display of sideslip. (Sideslip oscillations are often encountered during the reentry phase.)

The 26-degree angle-of-attack is maintained until the g-pointer is again observed moving toward the center of the null indicator, which now displays error with respect to the reentry level of 5.2 g. The display picture at the time of initiation of the roundout maneuver is shown in Figure 9-6. The vertical velocity pointer is shown indicating the relatively high rate-of-sink which characterizes the beginning of the roundout maneuver. The path can again be used to display altitude error if desired. Dynamic pressure and total velocity are monitored using supplementary indicators.

The roundout acceleration level is held until the vertical velocity pointer indicates the proper value of +300 feet per second for termination of the maneuver (in this case, the top of the scale could be used), whereupon the pilot reduces angle-of-attack to brake his rate-of-climb. This is done to reduce the chances of "skip-out" from the atmosphere where the pilot is unable to change course.

The pilot is next required to fly to high-key* which usually involves visual acquisition of the landing site. The

* High key is the initial checkpoint for the landing pattern, usually directly above the landing lake at about 30,000 feet.

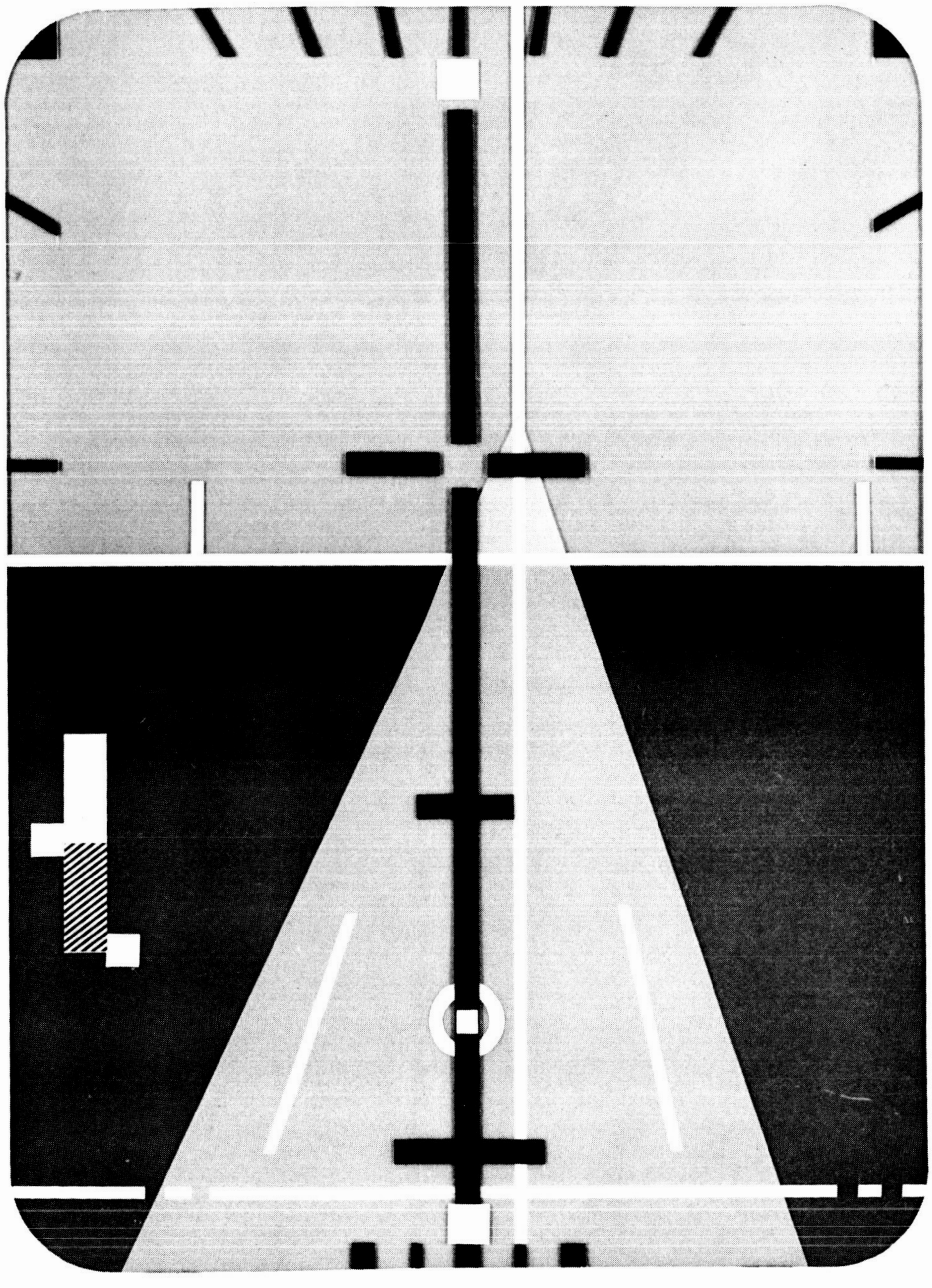

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Figure 9-6. Reentry Roundout

pathway could be programmed, however, to give navigational information for this task. Identifiable ground features are used for checkpoints during this phase. The pilot uses established energy-management procedures to arrive at the high-key point with the proper terminal conditions.

The landing phase is conducted according to a spiral landing pattern, requiring a 45-degree banked turn maintained for 360 degrees, during which altitude, airspeed, and position relative to the touchdown point are carefully controlled. At each 90-degree heading change, subsequent checkpoints are provided for altitude, airspeed, and position. The second of these (low-key) requires approximately 14,000 feet altitude, and the third requires 8500 feet altitude. The turn is terminated at 3500 feet and at approximately 2 miles off the end of the runway. A 1.5-g flare-out is initiated at 3000 feet indicated altitude (800 feet above the ground).* Touchdown occurs upon completion of the flareout. The flight terminates with a 5000 to 6000-foot slideout on the lake surface. Sideslip control is maintained by the pilot using the vertical stabilizer.

Throughout the landing pattern, the pilot depends heavily on visual cues for controlling vehicle rate-of-sink, relative position, and height above the ground. Normal perspective cues are not readily available during the final phase of landing due to the extent of the landing lake and the lack of external references. However, pilots indicate that the texture of the landing surface provides sufficient reference at low altitudes.

An alternative method for establishing a landing pattern

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* The landing lake is 2200 feet above sea-level.

is the use of a straight-in approach. This method is not normally used, due to the lack of adequate external references (as previously mentioned) and the resultant increased difficulty of control. This difficulty might be improved by the use of the integrated display with a navigational glide path similar to the one shown in Figure 9-7. Ground texture has been added, together with a ground position identifier (GPI) symbol, to furnish the necessary perspective cues. These elements will translate toward the aircraft in the display in proportion to the vehicle airspeed. By maintaining his velocity vector symbol on the GPI, the pilot will establish the proper flight path-angle and sink-rate. Airspeed information relative to a command airspeed can be displayed with a broken-line on the side of the path (which remains stationary for zero error) or, in the absence of this command information, tarstrips such as those mentioned in paragraph 8.2 can be used to display airspeed. The near-end of the glide-path indicates altitude error by changes in width, and positional error by displacement in the horizontal direction. Necessary inputs for such a path can be provided by ILS or TACAN navigational systems with suitable interface equipment.

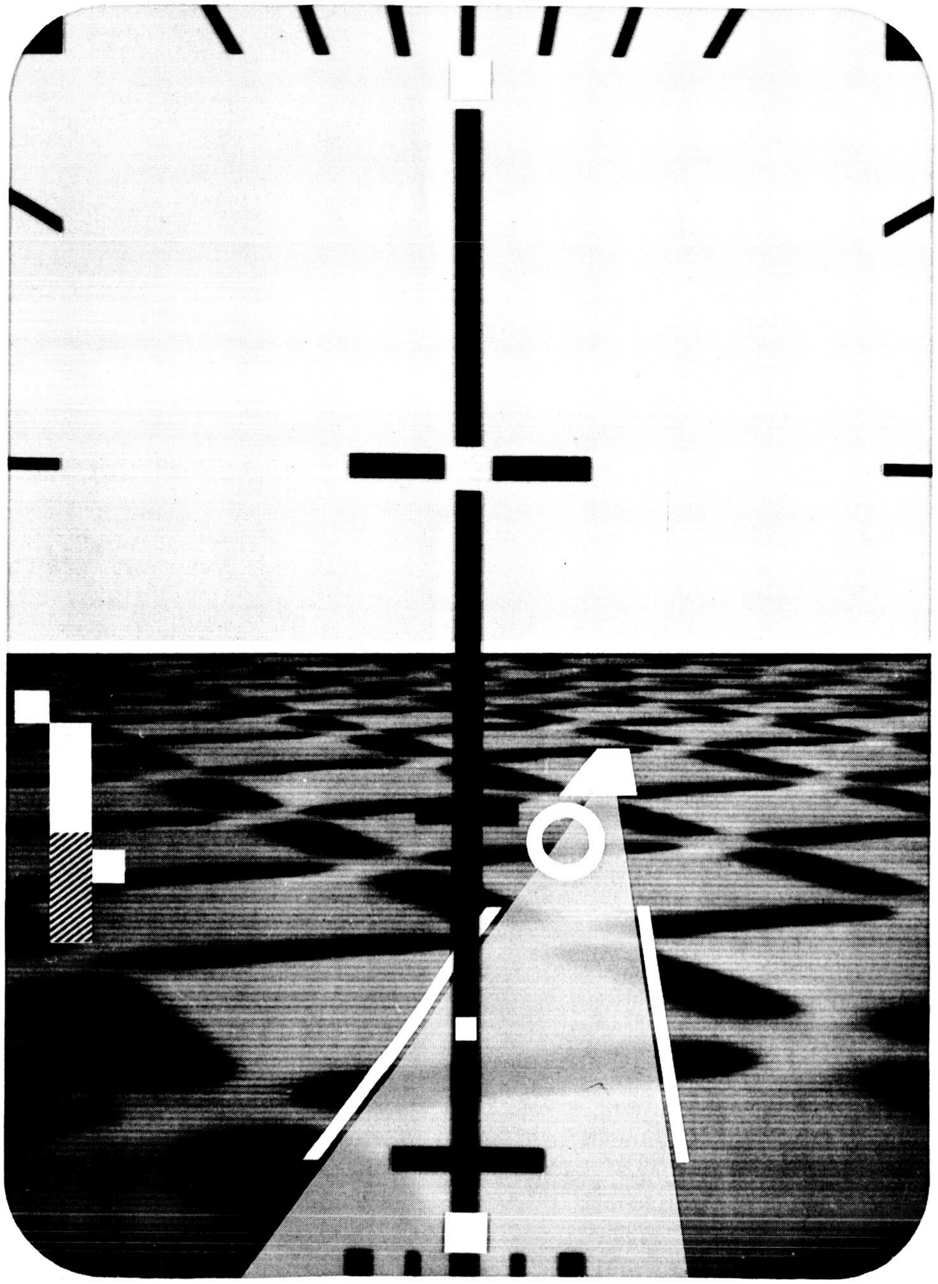

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Figure 9-7. Landing Approach

10. DISPLAY MECHANIZATION

10.1 Block Diagram

Figure 10-1 is a block diagram for one possible mechanization of the second generation integrated display. This display is generated entirely by standard and micro-size semiconductor circuitry, with the exception of the CRT in the TV indicator.

The elements of this display are described in Section 8. A detailed description of the geometry associated with those display elements which are most complex from the standpoint of generation (pitch and heading reference markers and the path) is presented in Appendix D. The signal inputs to the display generator are described in detail in paragraph 8.3.

The block diagram shows the display generator. The output of the display generator is a single composite video signal containing all picture information. This signal is sent to a TV indicator (Figure 10-2) of standard design, but of military quality.

The signal inputs from the vehicle computer are, essentially, the independent parameters of attitude, angle-of-attack, sideslip, etc., which the display is to present in graphic form. In the signal conditioner, these signals are appropriately attenuated, scaled, and switched. Derivation of the tangent functions of α and β are also accomplished in this block. The input data signals are distributed from the signal conditioner to the coordinate transformers and the symbol video generating blocks.

The synchronizer, in addition to providing the synchronizing and blanking components for the composite video waveform,

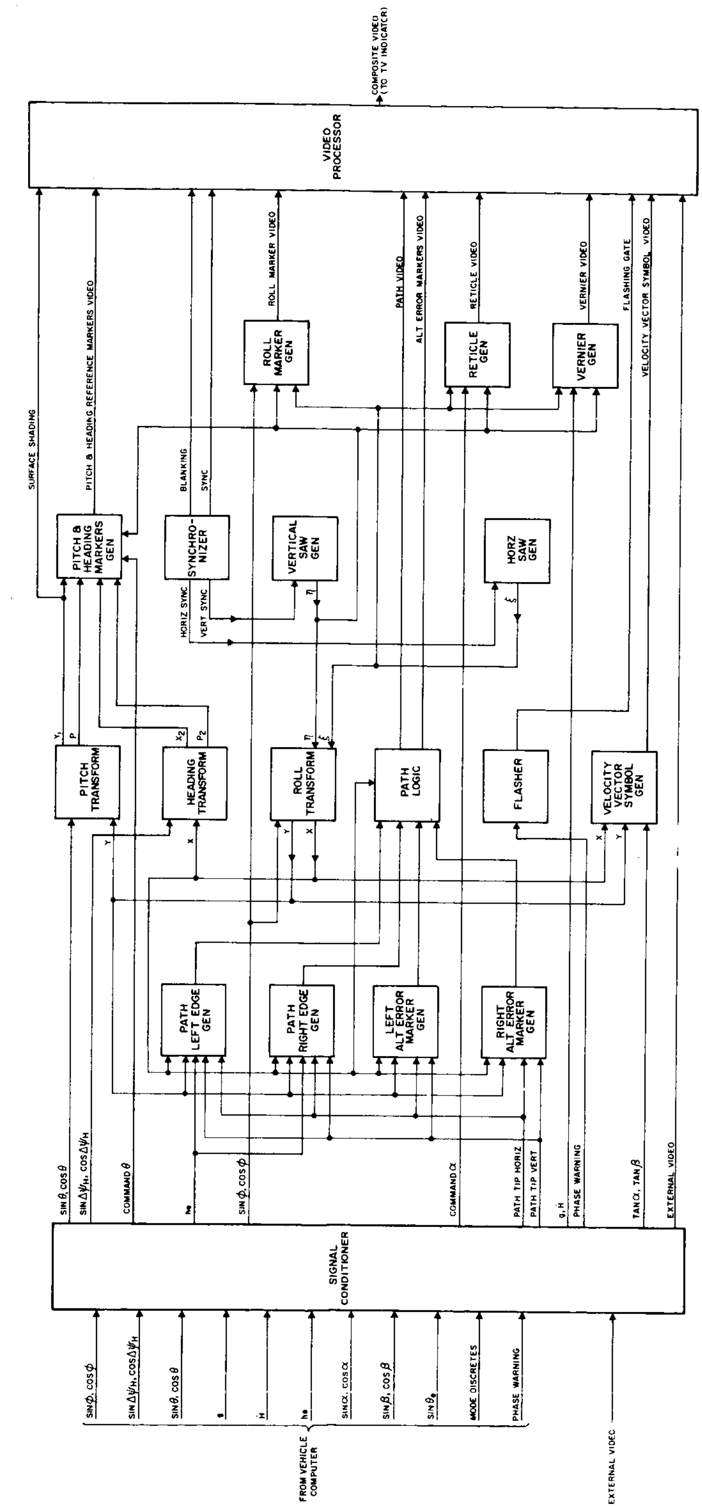

Figure 10-1. Circuit Block Diagram

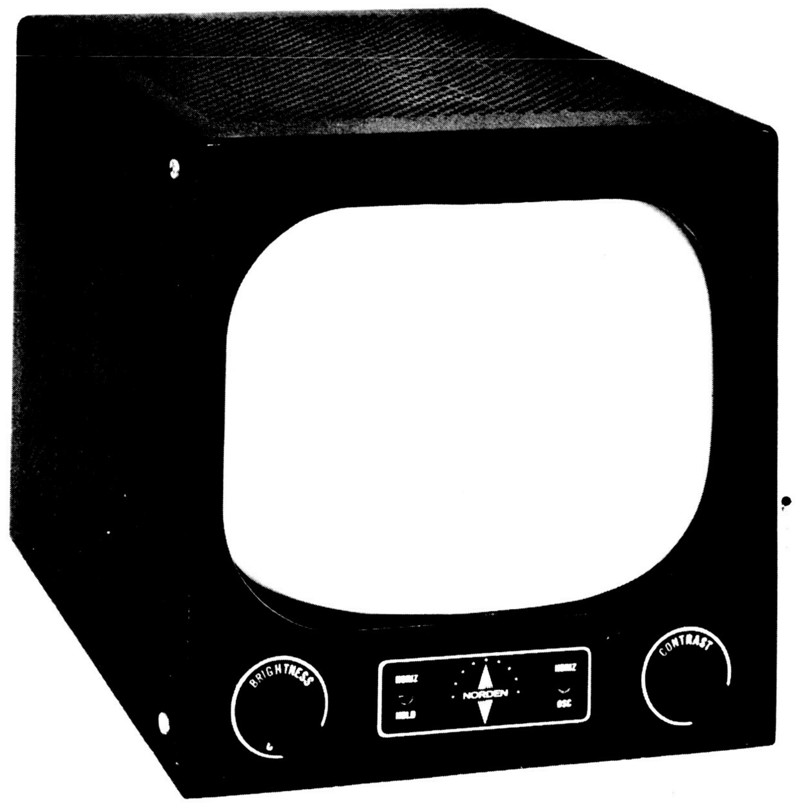

Figure 10-2. TV Indicator

provides the basic 60 cycle per second and 15 kilocycle per second timing gates which drive the screen coordinate saw generators. These signals are also used to synchronize the external TV camera when required.

The roll, pitch, and heading transformer blocks consist of analog summers and multipliers to compute the transformed coordinates x, y, y_1, p, x_2, and p_2 of the pitch and heading reference markers. Rolled screen coordinates x and y are also required for generation of the path and the velocity vector symbol.

Four blocks are allotted for the generationof lines which represent the edges of the path and the two altitude error markers. These are the solutions of equations for lines that join two points and are accomplished with summers, multipliers, and analog comparators. The four resulting gate signals representing the required lines are combined by the path logic block into two videos, one for the path and one for the altitude error markers.

Generation of the various symbols is divided into five functional operations. The computations associated with these operations utilize comparators, level detectors, and logic circuitry. The output of each functional circuit is a separate video signal for each symbol.

The flasher provides a three cycle per second gate signal for a period of 4 seconds from the time the phase warning pulse is received. The flashing gate is employed in the video processor to cause the reticle to flash for 4 seconds prior to a change in flight phase.

The separate video signals are combined in the video processor block, where they are modified for brightness and shading and selected according to the particular display mode to be presented. TV synchronization and blanking components are also mixed in here to form the complete composite video output signal.

10.2 Display Accuracy

The accuracy of a synthetic display is related to the difference between the actual screen position of a point in a displayed element and the position that this same point would occupy if there were no error in its placement, as defined by the independent parameter inputs to the display generator. For example, one measure of the accuracy of the velocity vector symbol (circle) is to set independent inputs α and β to zero, thus, theoretically putting the center of the circle on the center of the reticle. The circle observed on the screen will naturally deviate from this ideal position due to an accumulation of errors acquired in the circuitry which generates it. The mean of the linear distances from the center of the reticle to the center of the circle, measured a sufficiently large number of times and under various environmental conditions, is the mean error (E_M) under standard* display conditions. The quantity $\frac{E_M}{D}$ (where D is the screen diagonal) expressed as a percentage, is then the standard accuracy of the velocity vector symbol. If E_M were found to be 0.2 inch and D is 8 inches, the standard accuracy would be ± 2.5 percent.

To be thorough, the accuracy of a symbol should be determined under conditions other than standard (i.e., with the theoretical position of the symbol set at various points throughout the screen area), then the total accuracy for the symbol may

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* Standard display refers to the display that results when all the independent input signals which determine the position of a specified element are set at zero.

be expressed as the mean of the accuracies observed at each condition.

Since each picture element is generated separately, we expect standard and total accuracies to differ between elements. Obviously, estimating or measuring the accuracies of all picture elements in the display is an extensive process. In the absence of completely accurate data concerning a display, the standard accuracy of each of the critical display elements (the term <u>critical</u> applies to those elements of which the accuracies are essential to controlling the vehicle) should suffice.

The required accuracy of a display for the X-15 may be inferred from the accuracy of the presently used three-axis mechanical attitude instrument. This accuracy is ±0.5 degree at zero, and ±1 degree for all other angles with a viewing sensitivity of 33 degrees per inch. On the integrated display, one degree equals 0.116 inch yielding a viewing sensitivity of 8.6 degrees per inch. Therefore, ±1 degree required accuracy for all angles corresponds to ±1.45-percent total accuracy for the display, and ±0.5-degree required accuracy at zero corresponds to ±0.73-percent standard accuracy for the display. The acuity of the eye at 30 inches viewing distance is approximately 0.009 inch, or 0.08 degree on the display, so required accuracy is not limited by the visual capacity of the pilot. Also, a 525 line TV scan system will resolve the screen vertical down to 0.012 inch, so TV resolution does not limit the required accuracy either.

Present day analog-generated displays are capable of holding ±2.25 percent (±1.55 degrees) standard accuracies. This figure should be reduced below ±1 percent (±0.7 degree) using digital techniques. However, use of vernier rather than analog type indications does allow the display of critical parameters (angles) to ±0.75 percent (±0.52 degree) standard accuracies under present capabilities.

10.3 Display Response Time

The response time of the electromechanical flight instruments in the X-15 is in the range of from 0.25 to 2 seconds. The visual response time, or information rate, of the all-electronic integrated display (assuming a quick decay CRT phosphor such as the P4 or P31) is 16.7 milliseconds which is the TV vertical scan period. From the standpoint of the display observer, then, there is no perceptible information delay caused by the electronic display equipment.

This response time may be increased by interface considerations, such as the addition of low-pass filters in analog inputs or the necessity for conversion from digital inputs having low iteration rates.

10.4 Display Size, Brightness, and Resolution

The nominal distance from the pilot's eye to the instrument panel of the X-15 (and most other aircraft) is 30 inches. Human factors studies of viewers observing 525-horizontal scan line TV indicate that viewing distances of less than four times the vertical screen height cause undue fatigue, presumably because of the increased perception of video noise and interlace jitter on the screen. As a result, a fixed viewing distance of 30 inches and a maximum screen height of 7.5 inches is recommended. Human factors also indicate that the eye can effectively perceive only 15 degrees of view in the vertical plane without requiring motion of the head. For a 30-inch viewing distance, this is equivalent to 7.9 inches. When subjected to high g-forces, the eye will experience effective vertical fields-of-view which are reduced below 15 degrees. There are other reasons for limiting the maximum display size. When fatigue occurs, centrally located objects are more easily detected than peripheral objects. Also, the brightness and motion of an overlarge TV display will tend to attract all attention to itself and mask the relative importance of information contained on other panel instruments. For these reasons, a screen height between 6 and 7 inches is recommended.

Table 10-1 lists some common illumination and brightness levels encountered in aircraft and spacecraft. Assuming that the maximum illumination of 10,000 foot-candles will be encountered in the X-15 cockpit, a brightness of 1000 foot-lamberts is needed for an easily perceptible display. If protection from high ambient-light washout and reflections on the display is employed (in the form of a light hood in conjunction with a collimating optical screen, or a fiber-optic CRT faceplate), the display brightness may be reduced to 200 foot-lamberts. Whereas, the use of a fiber optic faceplate is presently an expensive solution to this problem; the collimating optical screen has proven to be a highly practicable means of eliminating off-axis reflections from the screen face-place. In spacecraft, it is assumed that the general illumination level of the cockpit will be held to the comfortable 50 to 100 foot-lambert range by filtering out strong light seeking entry through the windows. In this case, the display should develop 25 to 50 foot-lamberts brightness and would not require ambient-light protection. An added benefit of low brightness displays is the low power necessary for their generation.*

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* Refer to the discussion of display power-requirements in paragraph 10-5.

Table 10-1. Common Illumination and Brightness Levels

CONDITION	FOOT-CANDLES OR FOOT-LAMBERTS
Bright skies, maximum daylight	10,000
Bright skies, normal daylight	2000
Required for detail work	50 to 100
Brightness required for panel instruments	50
Required for prolonged work and reading	30
Normal home TV white-area brightness	25
Brightness of panel instruments for use with dark-adapted eyes (using red night filter)	10
Emergency lighting	3

The mean visual acuity of the eye under normal illumination conditions (50 to 100 foot-candles) is 1 minute of arc, or 0.009 inch at 30 inches distance. Assuming a screen height of 6.4 inches at 55 degrees and a required display accuracy of ± 0.5 degrees at ± 0.058 inches*, the pilot will be easily capable of reading his display to the required accuracy. 525 horizontal scan-lines will resolve the screen vertical to within 0.012 inch, which guarantees that the resolution of a 525 scan line system will provide the required accuracy.

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* Refer to complete discussion of display accuracy in Section 10.2.

10.5 Packaging and Mechanical Design

The recent history of aircraft electronic systems is characterized by a trend toward higher accuracy and complexity coupled with reduced weight, volume, and power. The fulfillment for these requirements is microelectronic packages and digital techniques. The use of microelectronic packages as thin-film and integrated semiconductor networks results in greatly increased reliablity and a significant reduction in the number of interconnections and connectors, since standard components are replaced by entire circuits. Fully-digitized techniques for coordinate transformation are not presently practicable, but will be available within two years.

Fifty percent of circuitry for present-day synthetic displays consists of non-microelectronic, conventional components mounted in high-density cordwood modules (see Figure 10-3). Certain modules also contain several presently-available integrated analog,microelectronic circuits such as sense amplifiers, comparaters, and a-c signal amplifiers mounted in transistor cans. Digital techniques account for approximately forty percent of present synthetic displays, and when digital circuitry is used, it may be implemented by Fairchild-type low-level logic mounted in cordwood modules. Each module is designed to perform a discrete electronic function.

The modules are generally arranged in rows on circuit cards (see Figure 10-4). Each card is designed to perform a discrete display function. The cards are stacked in a frame (Figure 10-5) which provides structural support, easy access, and adequate clearances for convection cooling by forced air.

Defective modules are considered expendable and non-

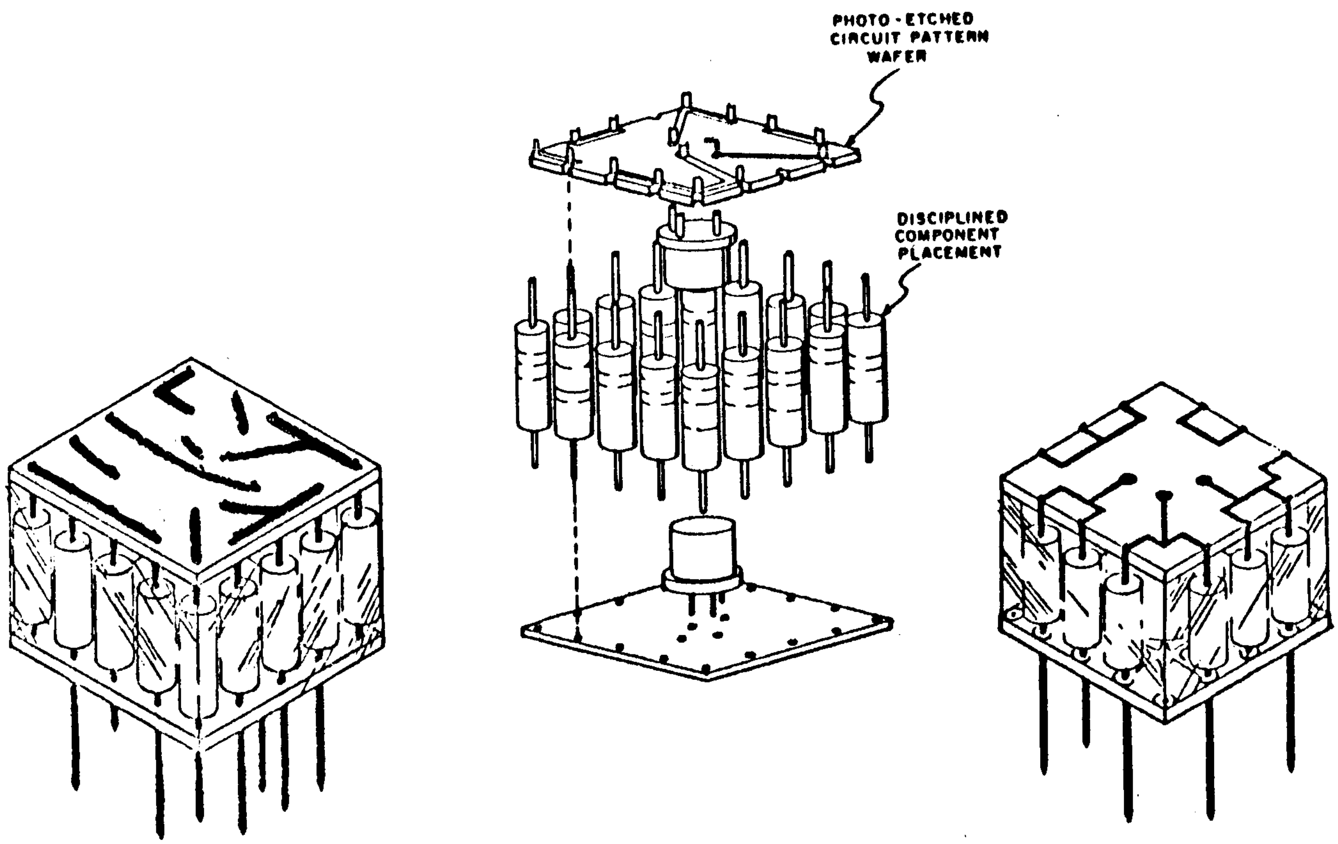

Figure 10-3. Typical Cordwood Electronics Component

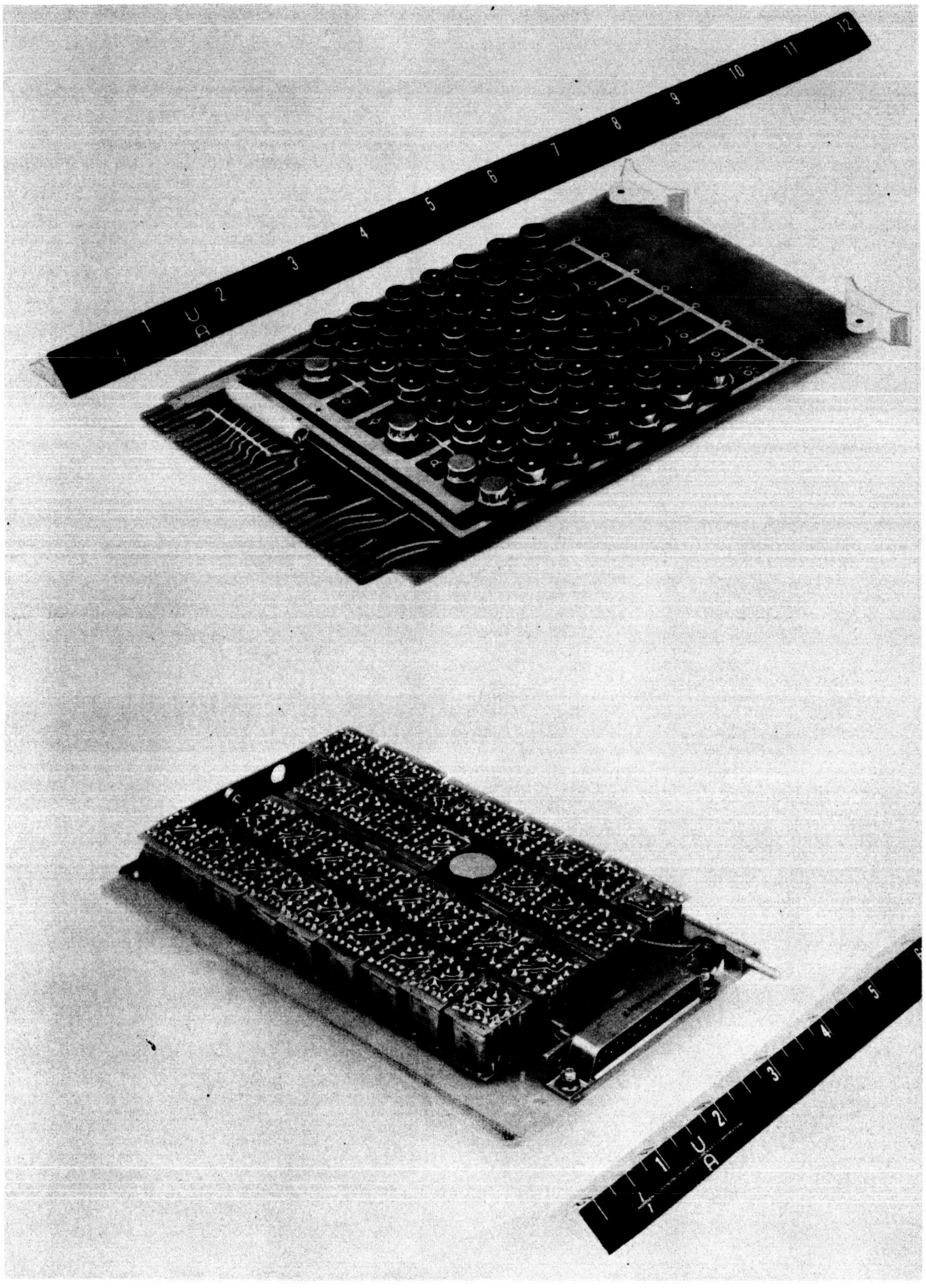

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Figure 10-4. Completed Circuit Cards.

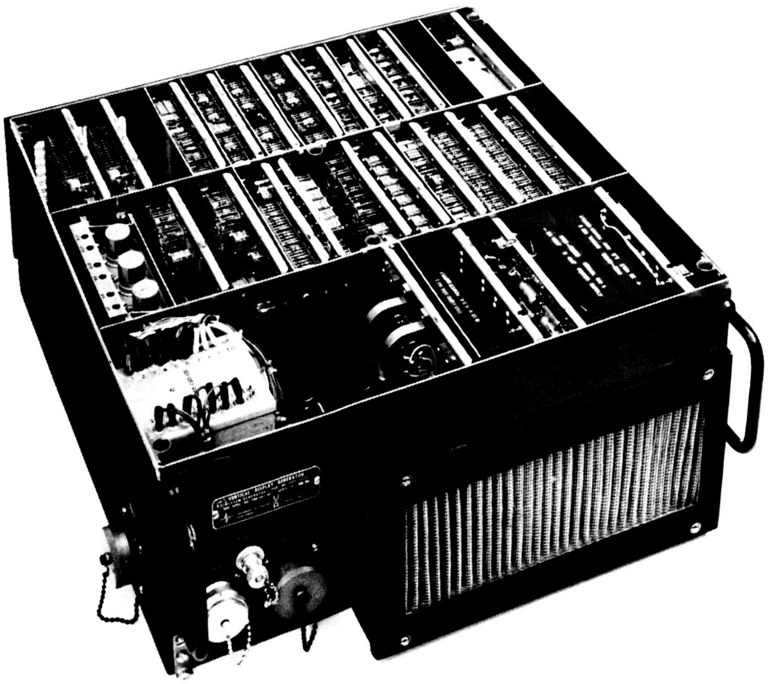

Figure 10-5. Circuit Card Frame

repairable. Cards which exhibit a malfunction are repaired by replacing their defective modules.

Adhering to this type of construction, the following characteristics have been calculated for the two units shown:

	DISPLAY GENERATOR	TV INDICATOR
Volume	730 cubic inches	730 cubic inches
Weight	25 pounds	25 pounds
Power	95 watts	70 watts
MTBF	1250 hours	3000 hours
Total MTBF	882 hours	

The over-all physical characteristics of the state-of-the art integrated display are illustrated in Figure 10-6. When, with further development, the full impact of microelectronics and digital techniques is realized, the figures stated above for the display generator should be improved by the following minimum reductions:

Volume reduction - 3 to 1
Weight per cubic inch reduction - 4 to 3
Power per electronic function reduction - 2 to 1

A monitor for use in a true spacecraft, where ambient illumination levels will presumably be controlled within the 50 to 100 foot-candle range, could be designed with approximately 250 cubic inches, 10 pounds, and 10 to 15 watts dissipation.

10.6 Reliability and Maintainability

The reliability of the state-of-the-art display system has been calculated to be approximately 882 hours MTBF. This figure

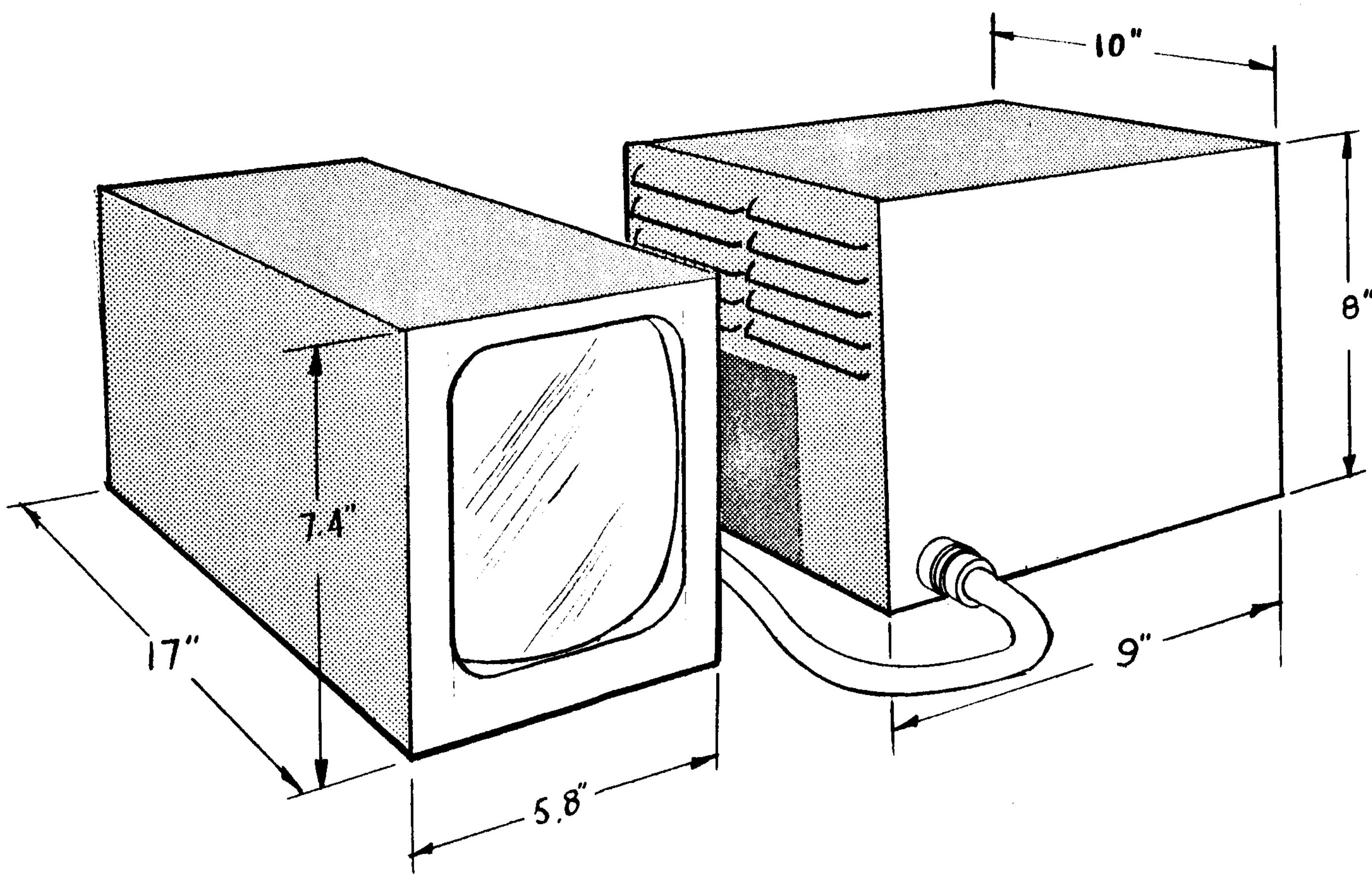

Figure 10-6. System Configuration

compares favorably with the 1000-hour MTBF specified for the present all-attitude indicator. With the advent of improved techniques, the display system of several years hence should attain a MTBF of at least 6500 hours. The CRT, the most important single component in all-electronic display systems, has a proven MTBF of approximately 33,000 hours (Section 15, reference 14). Alternative display devices which hold promise of increased reliability are described in Appendix E.

The maintainability of the display system depends on the following factors:

a. Modularization - Modules and circuit cards are grouped according to the display element or elements they affect on the screen, thus making a malfunctioning circuit or adjustment logically and easily identifiable.

b. Accessibility - All test points are available without removing circuit cards from their frame. Modules mounted on a circuit card are easily accessible by placing the circuit card on an extender card. All circuitry is located on circuit cards so that it is easily monitored or removed from the display system for repairs.

c. Standard Test Equipment - With the aid of a few standard items of commercial test equipment (oscilloscope, multimeter, digital voltmeter), all levels of maintenance can be performed by engineering personnel.

d. Documentation - Complete troubleshooting charts and alignment procedures greatly reduce maintenance time. The nature of the TV display format is such that many malfunctions can be isolated by observing the behavior of the displayed picture elements.

e. BITE (Built-In Test Equipment) - Although not a necessity, BITE is recommended for aircraft or spacecraft electronic equipment where in-flight maintenance or specific information about the usefulness or partially malfunctioning equipment are important features.

One portion of the BITE samples internal electronic functions and provides go no-go status signals or visual indications when the prime equipment is either out of calibration or exhibits a gross electronic malfunction. The status signals usually have the capability of isolating faults to a circuit card, group of cards, or particular circuit function. Sampling is performed automatically, continuously, or by an external command signal.

Another portion of the BITE is capable of replacing the input data signals of each functional group of circuit cards with a set of static test signals. The static test signals produce a static test display containing all display elements on the screen. Comparing this test pattern with an internally generated test reticle, the pilot or maintenance man is able to quickly ascertain the accuracy of the display elements. The test pattern is actuated manually.

Used in conjunction with the continuous automatic signal monitoring, the static test pattern gives better than 95 percent assurance that the equipment is operating within a specified performance level. BITE is a highly desirable feature which normally adds 10 to 15 percent to the size of the basic display equipment.

10.7 Summary of Characteristics

The physical and operating characteristics of the integrated display system are summarized in Table 10-2. The integrated display provides continuously available flight information to the pilot in terms of eleven important flight parameters. Mode-switching has been used only where necessary to reduce the possibility of misinterpretation.

Where command information is available throughout a flight, the Generation II display simplifies piloting tasks by presenting several parameters in the form of a single error indication.

The field-of-view in conjunction with the offset reticle allows display of velocity vector information over the entire range normally encountered in flight (refer to Table 5-1). Attitude parameters are provided with full 360-degree range in all three axes , although numerical readout is limited to ±60 degrees in pitch, ±90 degrees in roll and ±15 degrees in heading.

The viewing sensitivity is approximately four times greater than that provided by the all-attitude indicator, making the use of venier readouts for pitch and angle-of-attack unnecessary. Vernier-type indications can be used, however, to provide greater accuracy as described in paragraph 10.2.

The display response is more than adequate to accommodate parameters rates normally encountered in flight, allowing a large margin for degradation by interface requirements.

Table 10-2. Integrated Display Characteristics

PARAMETERS DISPLAYED	RANGES
Altitude error*	Variable
Angle-of-Attack	-10 degrees to +35 degrees
Angle-of-Sideslip	-15 degrees to +15 degrees
Climbout Pitch Angle	Zero degrees to +60 degrees
Cross Track Error*	Variable
Flight Path Angle	-60 degrees to +60 degrees
Heading Error	360 degrees
Normal Acceleration	Variable
Phase Warning	Variable
Pitch Angle	360 degrees
Reentry Angle of Attack	-10 degrees to +35 degrees
Roll Angle	360 degrees
Roll Rate	Variable
Vertical Velocity	Variable
Command Pitch Angle*	Programmed
Field-of-View	40 degrees horizontal
Accuracy	
Normal Indication:	±1.55 degrees
Vernier Indication:	±0.52 degrees
Response Time	16.7 milliseconds
Viewing Sensitivity	8.6 degrees per inch
Panel Area	5.8 x 7.4 inches = 43 square inches
Display Area	4.8 x 6.4 inches = 31 square inches
Volume(Display Generator)	730 cubic inches
Volume(Display Indicator)	730 cubic inches
Weight(Display Generator)	25 Pounds
Weight(Display Indicator)	25 Pounds
Power (Display Generator)	115 VAC, 400 cps, 95 VA
Power (Display Indicator)	115 VAC, 400 cps, 70 VA
Reliability	882 hours MTBF

* Generation II Display Only.

It can be seen from the table that the CRT indicator utilizes more than 70 percent of the necessary panel area for information display. The comparable figure for the attitude indicator is 56 percent (Table 7-2).

Panel depth required by the integrated display indicator is approximately twice that of the attitude indicator, unless indicator width is increased behind the panel. The weight of the indicator is approximately 40 percent greater than the combined weights of the separate indicators for the parameters concerned. Electrical power required is 10 percent greater than the combined indicator requirements.*

The physical and electrical characteristics of the display generator are comparable to those of the indicator unit with the exceptions that no panel area is required and input power requirements are slightly greater for this part of the system.

The reliability of the total integrated display system, expressed in terms of Mean Time Between Failures, is 882 hours, compared with the specified figure of 1000 hours for the present attitude indicator. Reliability figures for the other cockpit indicators were not available for comparison.

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* It should be noted that the information display area of the integrated display is 2.3 times that of the attitude indicator.

11. INSTRUMENT PANEL MOCK-UP

A mock-up of an advanced instrument panel configuration containing the integrated display is shown in Figure 11-1. In comparison with the present panel (Figure 7-1), it can be seen that the integrated display indicator is centrally located in the position normally occupied by the attitude indicator. On either side are located vertical moving-tape indicators for qualitative readout of important flight parameters, including (from left to right) dynamic pressure (q), normal acceleration (g), angle-of-attack (α), vertical velocity (V_v), altitude (H), and total velocity (V). These instruments offer improved readability over presently used indicators and offer a physical configuration compatible with the central display.

Mode indicating lights are located directly above the display indicator. These inform the pilot of changes in display-mode throughout the flight.

For presentation of navigational information, a horizontal-situation indicator is located directly below the integrated display. The curves shown represent glide range capabilities for various conditions of flight in the form of an overlay. Aircraft and landing site positions are computed and displayed in relation to these curves to give the pilot energy-management information.

Engine instruments have been dualized and relocated to provide the most efficient use of panel space. Controls for the Stability-Augmentation System have been moved to a panel position to the left of the horizontal display. The timer has been relocated to the right of the integrated display indicator.

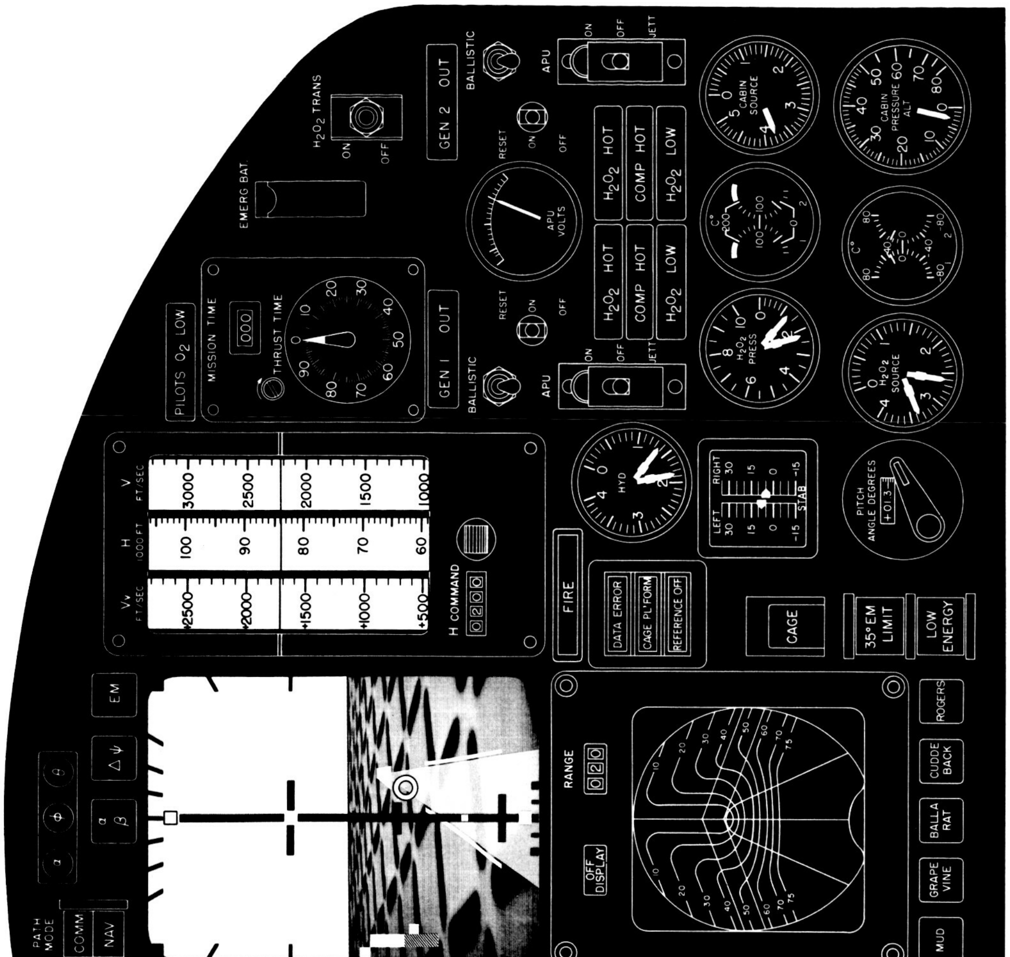

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Figure 11-1. Instrument Panel Configuration

The arrangement shown in Figure 11-1 provides a maximum of critical flight information within the pilot's central cone of vision and sufficient supplementary instrumentation for continuation of the mission in the event of failure of the integrated display indicator.

12. SIMULATOR STUDIES

During the course of this study, a program has been in progress at NASA Flight Research Center for investigating applications of contact analog display techniques to the control of hypersonic and other types of aircraft. The display equipment used for this program is shown in Figure 12-1.

A small TV monitor has been mounted on the panel of a general purpose cockpit simulator with supplementary vertical-tape indicators on each side. The ideo signal for the monitor is provided by the contact analog display generator shown on the right. This equipment contains the necessary circuitry for generating basic display elements and symbols, together with a 17-inch monitor and three power supplies. The cockpit is provided with a three-axis side-arm controller and rudder pedals for vehicle control.

Six-degree-of-freedom equations of motion are provided by two PACE 31R analog computers with an EAI TR-48 computer used for interface between the main computers and the display generator. Synchro resolvers are used to resolve angles into sine and cosine components. Command inputs are derived from an x-y plotter connected as a function generator.

The experimental display includes the elements shown in Figure 12-2. The ground plane is provided with several types of texture which can be selected for presentation, including grid lines, checkerboard pattern, and quasi-random texture. The sky texture consists of clouds appearing in true perspective. A

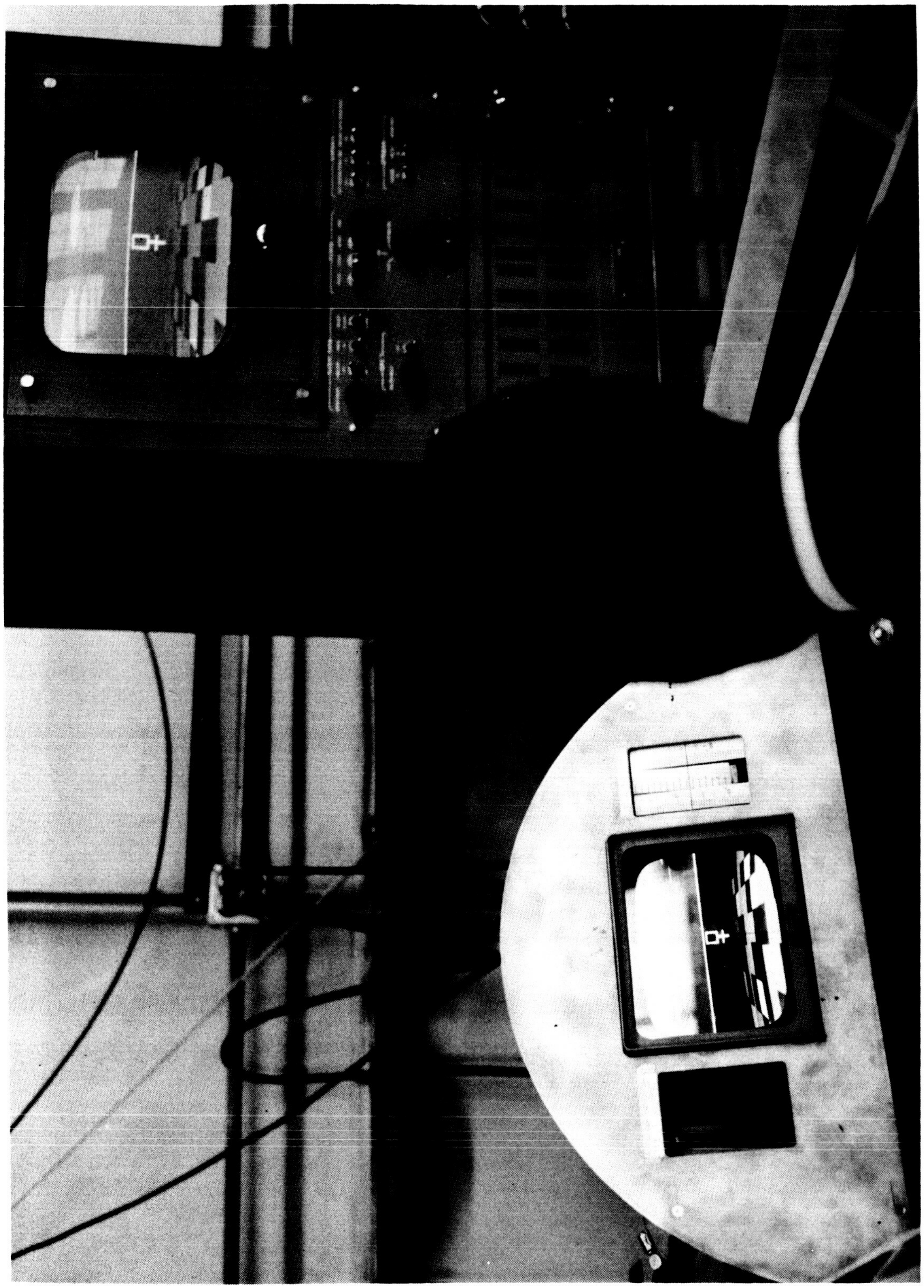

Figure 12-1. NASA Flight Research Center Simulator.

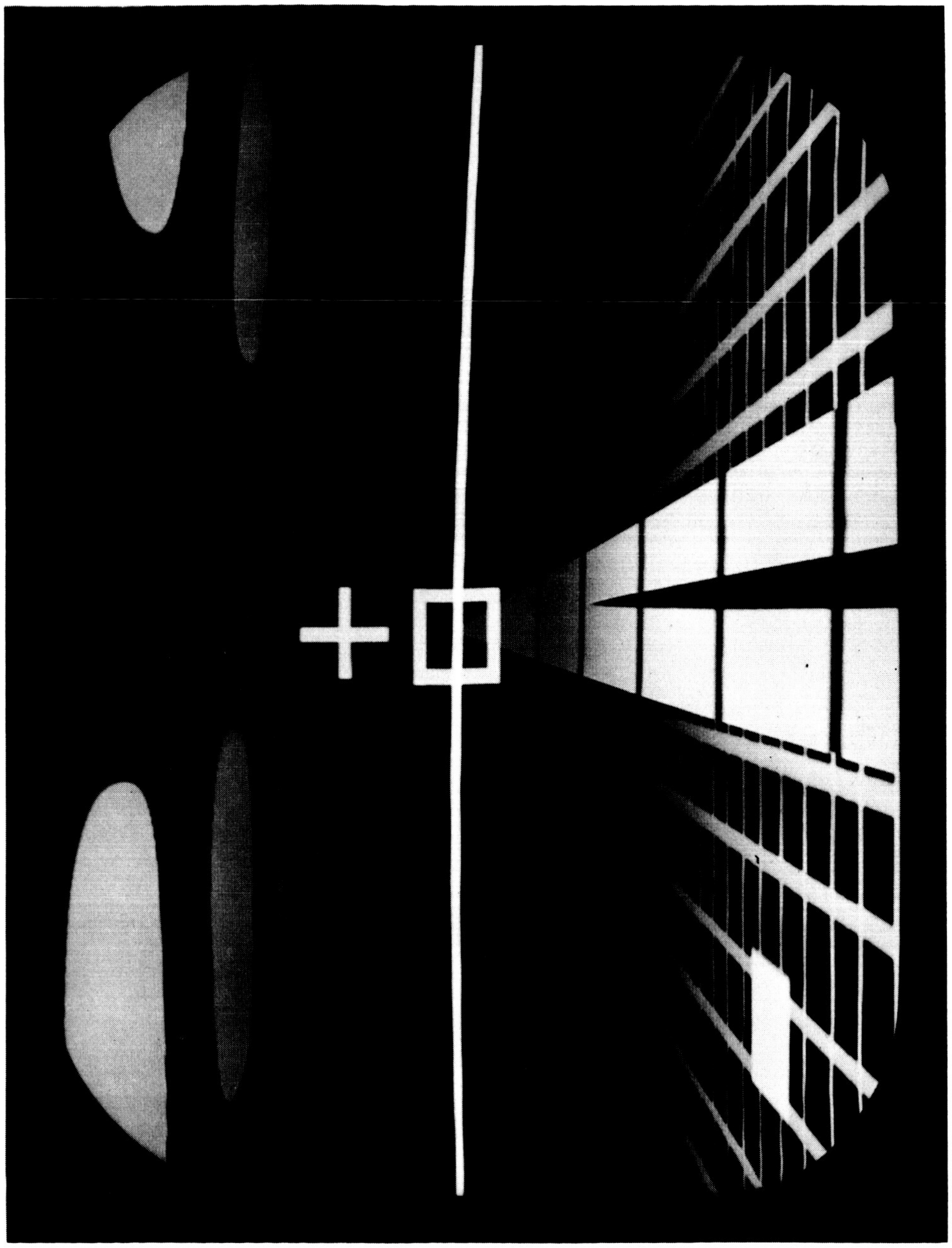

Figure 12-2. Contact Analog Display

white horizon line separates the ground and sky planes.*

Inputs are provided to vary the ground texture with altitude, ground track - velocity, and heading information in addition to the roll and pitch attitude inputs.

An open square and cross are available for use as fore-aft axis markers, velocity-vector symbols, quickening symbols, and representation of other non-pictorial parameters. A ground position identifier is also available for indicating ground checkpoints. The field-of-view represented by this equipment is 30-degrees vertical by 39-degrees horizontal.

A command path is provided containing a centerline and tarstrips which move toward the vehicle. Path orientation is described by roll, pitch, heading error, slope error, altitude error, and lateral offset. This system contains several modes of pathway presentation. These include command, navigational, and reentry modes. Various path configurations are shown in Figure 12-3.

In the command mode, the pathway is vehicle-stabilized, i.e., fixed to a point beneath the aircraft. During normal flight, the path is approximately 15 feet below the aircraft and vanishes at the center of the screen. Heading errors cause the path tip to bend left or right, and slope errors move the tip up or down. Roll angles cause a corresponding roll motion of the path. A lateral displacement will cause the near-end of the

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*No supplementary pitch, roll, or heading references are available with this display equipment.

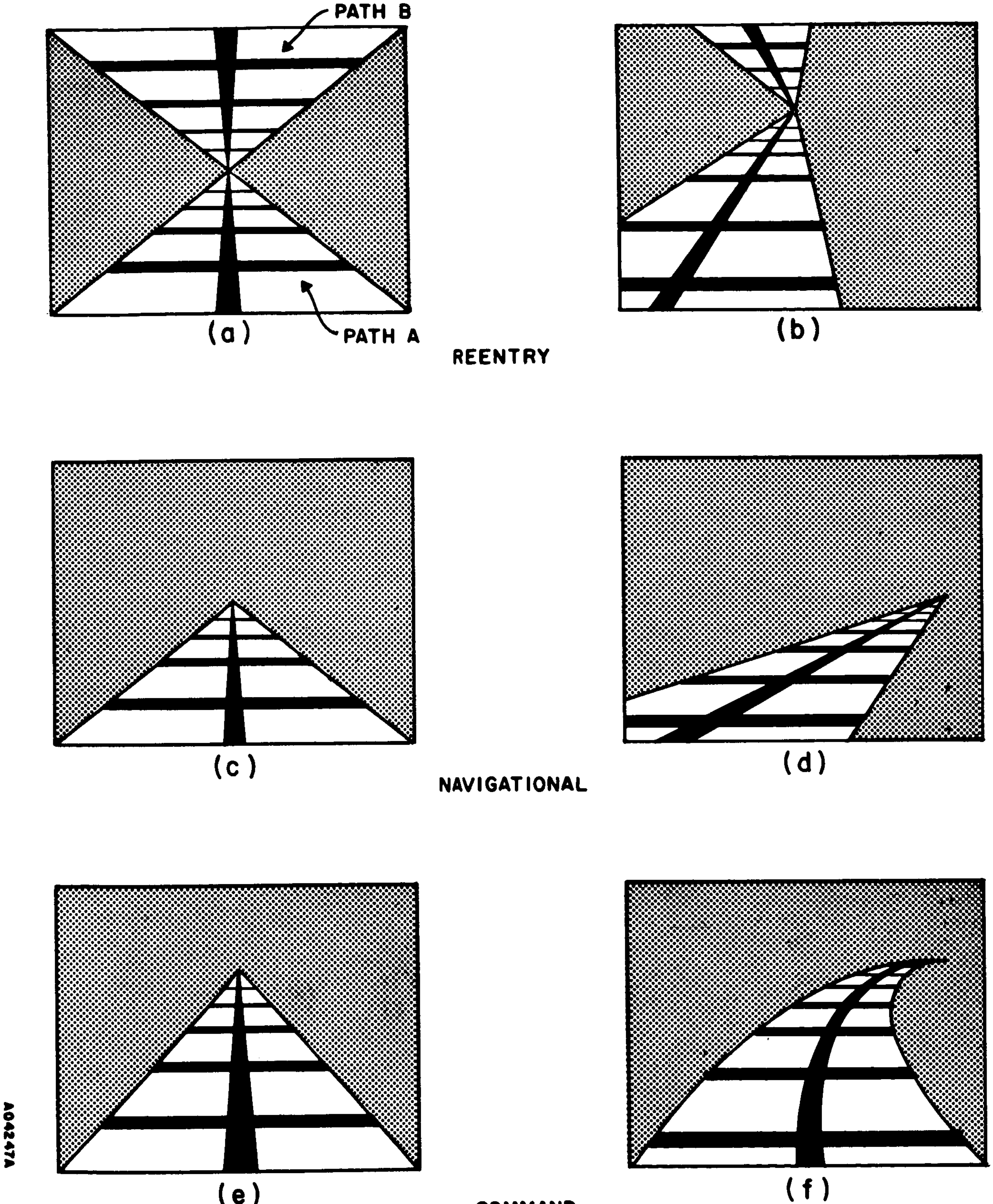

Figure 12-3. Path Configurations

path to move right or left while the tip remains fixed. The primary advantage of this mode is that the path does not disappear from the screen under conditions of large heading and slope errors.

The navigational mode is similar to the command mode except that the path is earth-stabilized. The path does not bend in this mode to indicate a command turn, but merely changes apparent heading. (As a result, excessive heading errors could cause the path to move off the screen.) This mode is used primarily in situations where input information is available in navigational form.

The reentry mode contains two paths which form a "corridor" for displaying upper and lower limits of flight, such as those defined by heating limits, deceleration limits, and minimum entry angles. The corridor has the same degrees of freedom as the navigational pathway.

Preliminary studies with this equipment have been concerned with the following general areas:

a. Evaluation of basic display elements for presentation of attitude, altitude, velocity, and specialized flight information
b. Techniques of command path programming
c. Application of display to X-15 flight phases
d. Application of display to other types of aircraft
e. Pilot reaction to display

Some of the specific areas under investigation are as follows:

a. Relative value of altitude error and velocity information displayed by the pathway

b. Path programming techniques derived from ILS and TACAN navigational aids
c. Improvements in pilot control using quickening techniques
d. Estimation of altitude rates during approach and landing
e. Generation of special symbols

Comparison of command path inputs are shown in Table 12-1.

United Aircraft Simulation Program

Simulator facilities have been established at major United Aircraft centers to aid in the evaluation of existing and developmental aircraft designs.

A view of the dual control simulator in use at the United Aircraft Research Center is shown in Figure 12-4. This system has been used extensively for the evaluation of contact analog displays for improved control of helicopters and other advanced VTOL aircraft designs.

Information obtained from this program has been an important factor in the human engineering phase of this study program.

Table 12-1. Typical Simulator Experiments

PATH PROGRAMMING METHOD	CONTROL TASK
Altitude error programmed on near-end of path	Follow programmed altitude profile using cues from path-width changes
Path tip programmed with flight path angle error	Follow programmed profile keeping path tip within aircraft axis symbol
Path tip programmed with flight path error minus angle-of-attack	Follow programmed profile keeping path tip within velocity vector symbol
Path tip programmed with altitude error plus altitude error rate	Follow programmed altitude profile keeping path tip within aircraft axis symbol

Figure 12-4. United Aircraft VTOL Simulator

13. CONCLUSIONS

An integrated display system has been presented which fulfills, to the maximum extent possible, the requirements of the basic design criteria listed in paragraph 8.1. These criteria are the result of a complete analysis of mission requirements and the application of human-engineering principles derived from earlier evaluations of related integrated display designs.

The limitations of existing integrated display methods were carefully investigated and improved where possible. New techniques were developed in those areas where present methods were inadequate. These areas include:

a. Use of reference markers for critical control parameters permitting improved quantitative readout and use of simplified ground and sky texture
b. Choice of aspect ratio and reticle location tailored to flight parameter requirements
c. Additional display capability derived from the use of null-type pointer scale indicators
d. Improved accuracy obtained by simplified computation methods

Some of the relative advantages and disadvantages of the proposed integrated display are summarized as follows:

A. <u>Advantages</u>

(1) The display provides increased information rate and reduced reaction time by consolidation of a diversity of flight data into a few symbols.

(2) Information ambiguity substantially reduced through the use of shape-coding and perspective presentation is provided.

(3) Flight director information for several parameters can be superimposed on situational display elements.

(4) The display possesses the capability for presenting additional data as new requirements are levied on the flight missions.

(5) Many types of data can be successively displayed through the use of mode-switching, providing increased utilization of display surface.

(6) Parallax problems are reduced by presenting all information in a single plane.

(7) The display has the capability of integration with closed-circuit television and windshield projection systems.

B. Limitations

(1) High reliability is required due to the concentration of several parameters in a single device.

(2) Means of protection from high ambient-light levels is required.

(3) Parameters having no counterpart in the real world cockpit view are difficult to present in contact analog form.

(4) Display value cannot properly be assessed due to the limited operational experience obtained to date with this type of display.

The high reliability necessary for integrated displays has been shown to be feasible with state-of-the-art designs. Display accuracy, although marginally adequate with present analog designs, shows promise of substantial improvement derived from digital computation techniques. Such design methods will also dramatically reduce display system physical requirements.

Further applications of this display method and other related display topics are described in Appendix E.

14. RECOMMENDATIONS FOR FOLLOW-ON PROGRAM

The techniques and devices described in this report provide a useful foundation for further developments in this field. Special areas of interest for future considerations are summarized in the following paragraphs.

14.1 Simulator Experiments

The answers to such questions as, "What is the optimum scale sensitivity?" or, "Are there perceptual conflicts between symbols in different modes?", can best be answered by evaluation of proposed displays in simulators and test vehicles. Such an evaluation may be conducted in the following three specific phases:

a. Evaluation of display content and format as a flight instrument
b. Observation of pilot performance with the display, applied to control of a simulated vehicle
c. Airborne evaluation of display

The first phase requires analysis of the general display concept in terms of one or more specific display formats. Such an analysis may be conducted with the use of several known criteria for evaluation. Some of these criteria are as follows:

a. Total Information Content - Content is analysed in terms of the number of parameters represented versus the number of symbols and total area used.
b. Scaling - Input sensitivities of each parameter is determined (where unspecified) and analyzed in terms of obtaining the required ranges with the required sensitivities.

c. Accuracy - Display indications are compared with input conditions and the resulting discrepancies are measured.

d. Quantitative Information Content - Where measurements of display indications are not possible, a subjective analysis is conducted to determine information content for each parameter.

e. Readability - Again, a subjective study is made to determine operator resolution of static display elements. Information overlap is also investigated.

f. Ambiguity - Symbols are analyzed subjectively in terms of ease of identification under changing modes of operation.

The second phase of evaluation involves the study of pilot performance using selected display formats. In the case of the format recommended in this report, such an evaluation would be conducted with a simulation of the aircraft for which the display was designed. Specific areas of investigation are as follows:

a. Attitude Control - The ability of the pilot to assume and maintain attitudes is measured under both preplanned and display-commanded flight conditions.

b. Recovery From Unusual Attitudes - Display content and behavior is analyzed in terms of the time necessary for the pilot to detect and correct unusual attitude changes.

c. Maneuvering Flight - A large amount of investigation is required to determine optimum methods of programming display elements for maneuvering flight. Included in this area would be the following:

(1) Evaluation of display during critical flight phases in terms of information content, accuracy of control, and pilot response time
(2) Selection of parameters for display of command information and amount of command information to be displayed
(3) Improvements afforded by the application of quickening, and prediction techniques to path and other symbols
(4) Ability to display navigational information derived from ground-based navigational aids
(5) Continuity of display during mode changes
(6) Determination of supplementary instrumentation required for completion of mission

Following a simulator evaluation of the display, it is recommended that an airborne evaluation of a specific display design be conducted. Such a program might involve several phases, culminating in the testing of a flight-rated system in a future X-15 mission. Preliminary airborne tests would be useful for finalizing the display format and for determining areas of improvement in initial flyable designs. The effectiveness of related display concepts, such as closed-circuit television or windshield projection, could also be evaluated.

Considerable experience in the design of airborne display equipment has already been derived from the continuation of a similar evaluation program pertaining to fixed and rotary-wing aircraft in the ANIP (JANAIR) display integration program.

14.2 Improvements of Display Limitations

Further attention should be devoted to the subject of extending the performance of existing system designs through the development of presently-available techniques. The most promising areas of attention are as follows:

a. Reduction of system volume through the application of digital circuit designs with microcircuit elements and high-density packaging techniques

b. Improvement in quantitative information content by the use of advanced techniques, such as alphanumeric symbols, bargraph indicators, video moving-tape indicators, and improved ground texture (App. E, Para. 4)

c. Further development in the field of solid-state display elements with the ultimate goal of generating a flat display medium (App. E, Para. 2)

d. An extensive study of the reliability of existing ruggedized CRT designs to determine the capabilities of available devices and requirements for additional development

14.3 Applications to Other Missions

The application of the advanced integrated display techniques discussed in this report to other types of missions involves conducting a thorough study of the mission and control requirements of the vehicle concerned, selection of a display format for the mission, and evaluation of operator-display performance in a manner similar to that described in paragraphs 14.1 and 14.2. Where similarities exist between missions, extensions of existing display concepts would suffice. This would be the case, for example, if the recommended X-15 display were to be applied to pilot control of boost, orbital insertion, and reentry of an orbital vehicle.* Other phases having no counterpart in X-15 flights, such as orbit correction, rendezvous, and lunar landing, would require more extensive consideration. A large amount of information has been gathered in the course of this study which would be useful in investigations of this type.

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* A possible format for an advanced reentry display is presented in Appendix E, Paragraph 5.

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APPENDIX A

A. X-15 VEHICLE CHARACTERISTICS

A.1 General

Length:	50 feet
Wingspan:	22 feet (wing sweep, 25 degrees)
Height (with lower vertical):	13 feet
Power Plant:	YLR-99 engine
Maximum Thrust:	60,000 pounds
Minimum Thrust:	24,000 pounds
Fuel and Oxidizer:	Anhydrous ammonia - liquid oxygen
Weight:	
Launch:	32,800 pounds
Burnout:	15,000 pounds
Payload (for experimental equipment):	2000 pounds
Electrical System:	115 vac 400 cycles per second
	26 vac 400 cycles per second
	28 volts d-c

A.2 Performance Characteristics

	DESIGN LIMITS	ATTAINED TO DATE
Altitude	250,000 feet	354,200 feet
Velocity	6,600 feet per second	6020 feet per second
Maximum Skin Temperature	1200°F	1150°F
Maximum Dynamic Pressure	2500 pounds per square foot	2000 pounds per square foot

APPENDIX B

B. INERTIAL PLATFORM INTERFACE DATA

FUNCTION	TYPE OUTPUT	SCALING
Heading	Synchro	1 degree per 1 degree shaft rotation
Pitch	Synchro	1 degree per 1 degree shaft rotation
Roll	Synchro	1 degree per 1 degree shaft rotation
Altitude	Synchro	500,000 feet per shaft rotation
N-S Velocity	Potentiometer	7000 feet per second per 2 K-ohms
E-W Velocity	Potentiometer	5000 feet per second per 2 K-ohms
Vertical Velocity	Potentiometer	5000 feet per second per 2 K-ohms
Total Velocity	-------------	Amplifier in indicator sums V_{NS}^2, V_{EW}^2, and V_H^2 and extracts square root through potentiometers
N-S Distance	Synchro	720 nautical miles per shaft revolution
E-W Distance	Synchro	240 nautical miles per shaft revolution

APPENDIX C

C. X-15 FLIGHT REQUEST

Flight No.: 3-22-36 Scheduled date: August 22, 1963

Pilot: Joseph Walker

Purpose: Expansion of ventral off reentry investigation, Altitude Predictor checkout, Photometer and Barnes Spectrometer experiments.

Launch: Smith Ranch on magnetic heading 170°, MH-96 Adaptive, Roll hold "ON", R.C. "AUTO", BCS "ON", heading vernier to "$\Delta\Psi$ ", ventral off.
Launch Point coordinates: 39° 30'N 117° 10'W

Item	Time	Alt.	Vel.	α	q	Event
1.	0	45	790	2	145	Launch, light engine, increase to 100% T. Rotate to 2g.
2.	11	45	1280	13	380	2g - Maintain 2g until $\Theta = 48^{\circ}$.
3.	29	56	2800	12	450	$\Theta = 48^{\circ}$ - Maintain $\Theta = 48^{\circ}$.
4.	84.5	176	5380	9	19	Burnout. Shutdown at a predicted altitude of 360,000 feet if burnout has not occurred and velocity is greater than 5200 fps.
5.	88	185	5350	10	15	Heading vernier to "STANDBY", check $\beta = 0$, then return to "$\Delta\Psi$ ".
6.	160	330	4350	-	-	Pushover to $\Theta = 0^{\circ}$. Roll to 45° bank angle, maintain 45° to peak altitude.
7.	196	360	4150	-	.16	Peak altitude. Roll wings level. Extend speed brakes to 20°. Maintain $\Theta = -20^{\circ}$ until approximately 220,000 feet, then establish $\alpha = 26^{\circ}$ for reentry.

Flight No.: 3-22-36

Item	Time	Alt.	Vel.	α	q	Event
8.	298	200	5250	26	6	Roll hold "OFF", heading vernier to "STANDBY".
9.	328	95	5200	26	550	5.2g - maintain 5.2g. Retract speed brakes when $\dot{H} \approx -500$ fps. (Max. reentry $q \approx 1200$ psf).
10.	353	70	3700	12	990	$\dot{H} \approx +300$ fps ($\theta \approx 18^\circ$) pushover to $3^\circ\alpha$. Max. $\dot{H} \approx +600$ fps. Vector to High Key. Reaction control "OFF". Engine Master "OFF".
11.	420	80	2800	5°	345	China Lake. Speed brakes as required.
12.						High Key - check flap and "SQUAT" circuit breakers in.
13.						Before APU shutdown, cycle flaps, set stabilizer, trim to zero, turn all data "OFF".

NOTES:

1. θ vernier will be set at 48°, a crosspointer will null at 26°.
2. Emergency lakes: Mud, Grapevine, Cuddeback.
3. Flight duration: Approximately 12.5 minutes.
4. Flight plan based on 59,200# thrust engine (50% = 34,900#).
   Total burn time at 100% = 84.5 sec.
   Launch wt. = 33,000#, Burnout wt. = 15,000#.

APPENDIX D

D. ANALOG SYMBOL GEOMETRY

Generation of a plane figure or symbol in true perspective on a TV display is accomplished in the following four steps:

a. Generation of functions which describe in real time the TV screen coordinates being illuminated by the electron beam (These functions are the TV sweep voltages.)

b. Transformation from screen coordinate functions to functions which describe the coordinates of the plane in which the figure or symbol exists. (The resultant functions are combinations of linear and hyperbolic functions that vary at the TV sweep rates.)

c. Continuous trial solution of the equations of the symbol. (The terms of the equations include the functions calculated in step b and independent parameters of aircraft attitude, altitude, etc., which determine the position and dimensions of the symbol in its own coordinate system. The number of equations required for a symbol is equal to the number of continuous curves which contribute to its shape.)

d. Generation of symbol video under conditions of exact solution of the symbol equations (This implies a comparison, which is the invariable requirement for equation solution and video generation.)

These principles of symbol generation are extended to many difficult display elements, such as:

- True perspective ground planes with 6 degrees of freedom
- Straight, curved, and segmented flight paths
- A variety of navigational, situation, and command symbols positioned in the picture or in any other arbitrarily oriented plane

Coordinate transformations are required for those symbols which appear in perspective. It is often possible to simulate the desired effects by approximating the equations of transformation or by limiting or removing various degrees of freedom. These approximations tend to result in a simpler, smaller, more accurate display generator, and they are justified where they do not limit pilot performance.

The orientation of the pitch and heading reference markers in the X-15 display are dependent on ϕ, θ, and Ψ as defined in Table D-1. Transformations from screen coordinates to the planes of these markers are described by equations which are derived directly from the geometry of the situation. The roll transformation equations are:

$$\gamma = \eta \cos \phi - \xi \sin \phi \quad \text{(rolled vertical)}$$

$$x = \eta \sin \phi + \xi \cos \phi \quad \text{(rolled horizontal)}$$

where η and ξ are the screen vertical and horizontal coordinates. The transformation which follows roll is pitch (Figure D-1) and the equations are:

$$\gamma_1 = d \sin \theta + \gamma \cos \theta \quad \text{(pitched vertical)}$$

$$\rho = d \cos \theta - \gamma \sin \theta \quad \text{(pitched viewing distance)}$$

It is easier to compute and use the scaled-down coordinates γ_1 and

Table D-1. Definitions of Symbols and Terms

SYMBOL OR TERM	DEFINITION AND MATHEMATICAL EQUIVALENT
Longitudinal Vehicle Axis	Datum axis running through the vehicle center-of-gravity in the fore-aft direction
Lateral Vehicle Axis	Datum axis through the vehicle center-of-gravity in the transverse direction perpendicular to the longitudinal axis
Vertical Vehicle Axis (Normal Axis)	Forms a right-hand orthogonal system with longitudinal and lateral axes
Ground Plane	Plane tangent to a theoretically smooth earth surface at the intersection of the earth surface and a radial from the earth center to the aircraft center-of-gravity.
Viewing Point	A point on the perpendicular to the geometric center of the viewing screen, spaced from the screen by the nominal viewing distance d
Screen Horizon	The intersection of the viewing screen and a plane parallel to the ground passing through the viewing point
η	The coordinate of a point on the viewing screen on the axis perpendicular to the vehicle lateral and longitudinal axes, measured from the screen geometrical center, positive up

Table D-1. Definitions of Symbols and Terms (continued)

SYMBOL OR TERM	DEFINITION AND MATHEMATICAL EQUIVALENT
ξ	The coordinate of a point on the viewing screen on the axis parallel to the vehicle lateral axis, measured from the screen geometrical center, positive right
ϕ	The roll angle of the vehicle; the angle formed by the lateral axis of the vehicle and a true horizontal in a plane perpendicular to the longitudinal axis of the vehicle
y	The coordinate of a point on the viewing screen along the axis perpendicular to the screen horizon, measured from the geometrical center of the screen, positive up
x	The coordinate of a point on the viewing screen along the axis parallel to the screen horizon measured from the geometrical center of the screen, positive to the right
θ	The pitch angle of the vehicle longitudinal axis from a true horizontal plane, positive values indicating climb
Ψ	The true heading of the vehicle measured from true north, positive values clockwise

SYMBOL OR TERM	DEFINITION AND MATHEMATICAL EQUIVALENT
Pathway	The pathway is a plane surface with essentially no thickness and finite width, and the length is considered to be infinite. To present information about linear and rotational positions, the pathway is capable of six degrees-of-freedom - three which define its linear displacement from an origin, and three which define its angular orientation with respect to a reference. When viewed on the display, the pathway surface appears as a perspective view.

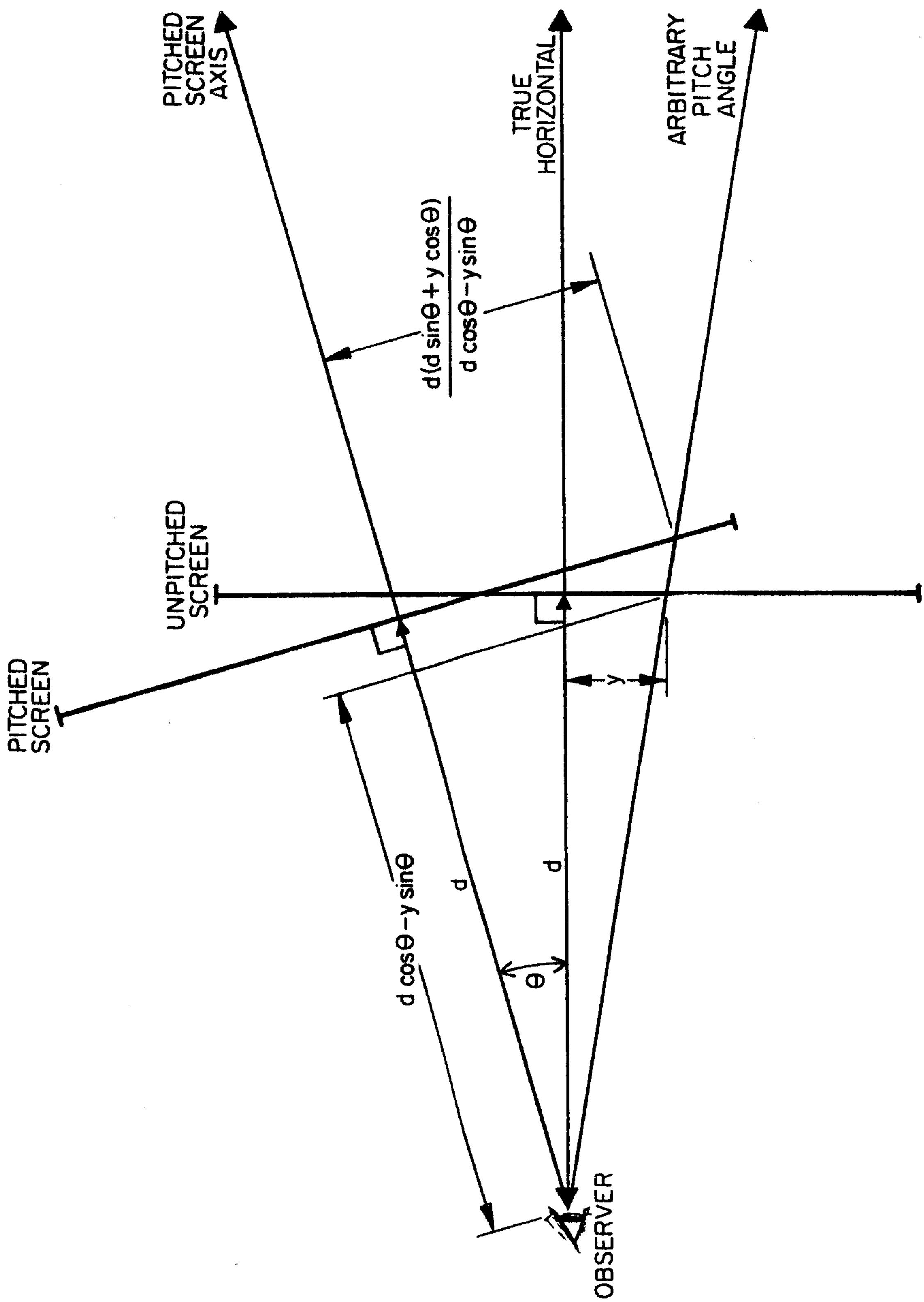


Figure D-1. Transformation Geometry

p for pitch rather than the true pitched coordinates $\frac{d\gamma}{p}$ and d. Similar to the geometry of the pitch transformation, the heading transformation is described by the equations:

$$x_2 = p \sin \Psi + x \cos \Psi \text{ (headed horizontal)}$$

$$p_2 = p \cos \Psi - x \sin \Psi \text{ (headed viewing distance)}$$

The result of these three transformations is the screen coordinate system p_2, γ_1, and x_2 which has been rolled, pitched, and headed.

From the many possible methods for developing the geometry of a pathway, the one chosen for the X-15 display is shown in Figure D-2. The path tip is deflected from screen center by θ_e and $\Delta\Psi_H$ according to the equations:

$$\eta_T = d \tan \theta_e \cong d \sin \theta_e \text{ (path tip vertical)}$$

$$\varepsilon_T = d \tan \Delta\Psi_H \cong d \sin \Delta\Psi_H \text{ (path tip horizontal)}$$

For a path that rolls, the near-end of the path is described by two points in x and y coordinates and located on the path edges. γ_A, x_A, and x_B are constants arbitrarily chosen to give the path a nominal width for zero altitude error ($h_e = o$). Quantity h_e is then employed as a lateral deflection on x_A and x_B in opposite directions to show altitude error. As seen from the geometry, the equations for the lines of the path edges reduce to the equations for lines joining two points in a plane (after the x and γ points have been transformed into η and ξ coordinates). The lines representing the altitude error markers are represented by similar equations, but are arbitrarily limited to the rolled vertical region between γ_B and γ_C.

The equations of the pathway lines and the equations of the pitch and heading reference markers are mechanized in high-speed

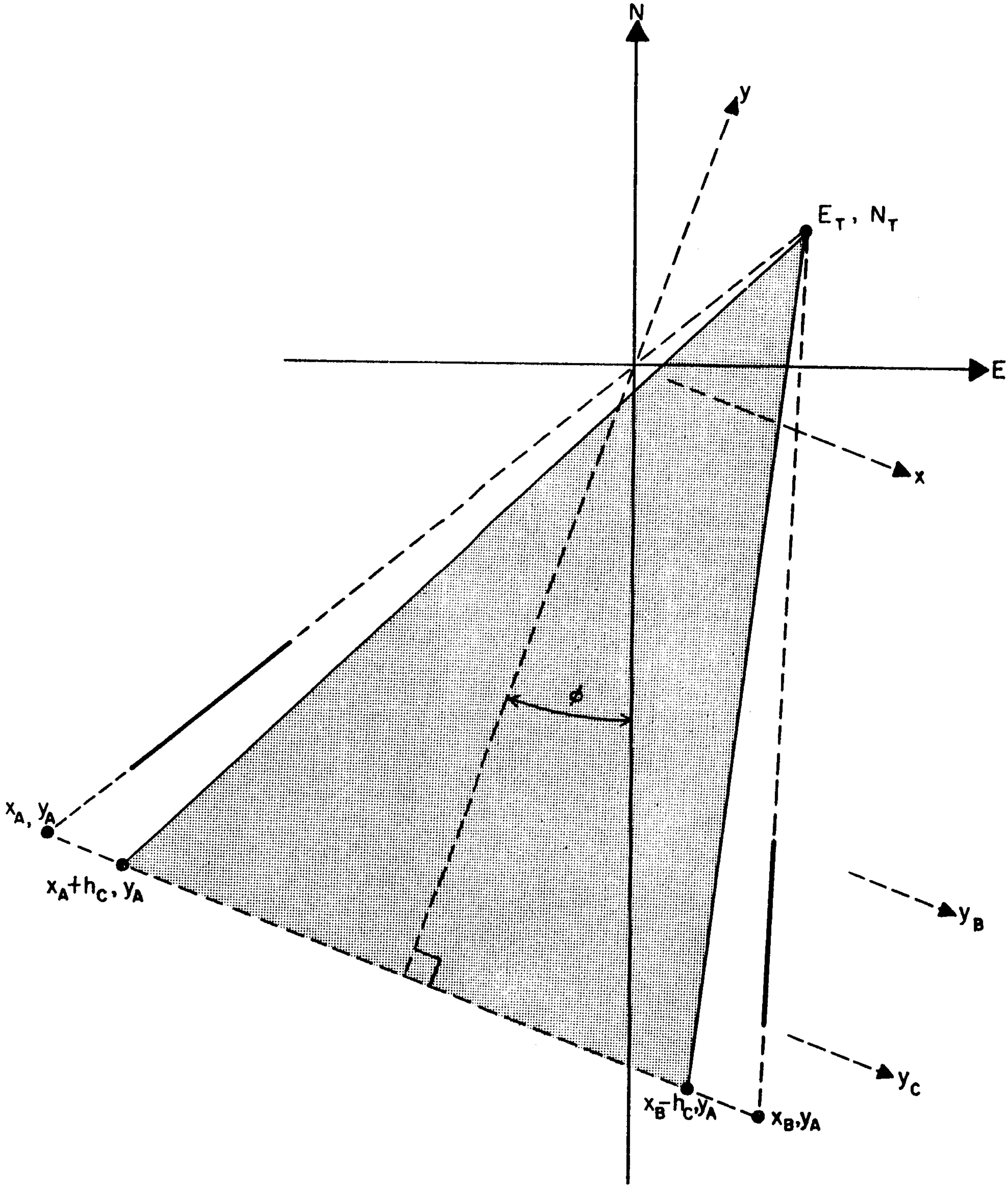


Figure D-2. Pathway Geometry

analog computing functional blocks which perform steps b and c, transformation and continuous trial solution.

APPENDIX E

E. EXTENDED DISPLAY CAPABILITIES

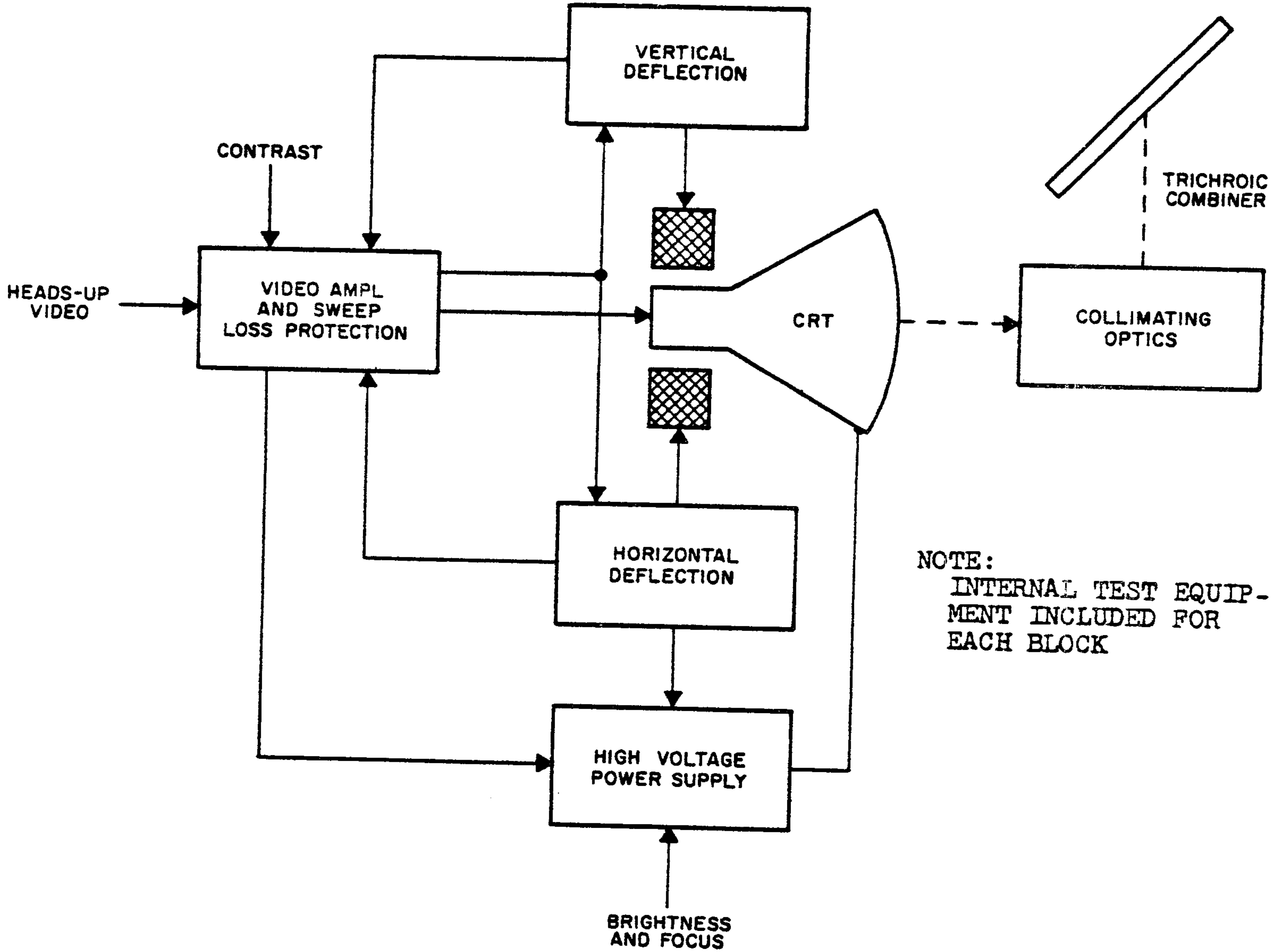
The purpose of this appendix is to describe possible extensions of the integrated display concepts presented in this report. These techniques have resulted from recent developments in the field of display design reported in the literature and resulting from laboratory developments which have taken place during the period of the study.

E.1 Cockpit Projection

Selected elements of the integrated display can be presented to the pilot in the form of a collimated image optically superimposed on the real-world view through the windshield of the vehicle. This technique is commonly referred to as "heads-up display".

Such a system normally consists of a small projection CRT, a system of collimating optics, and a special combining glass (See Figure E-1).

A new heads-up display concept is currently under development which provides improved performance with substantial reduction in system size, weight, and complexity as compared with earlier designs. This system utilizes a specially designed parabolic combining glass with a special 2-inch high-brightness CRT mounted at the focus. This system effectively combines the functions of the collimating optics and combining glass in one device, resulting in the elimination of the collimating optical package shown in Figure E-1.



NOTE:
INTERNAL TEST EQUIPMENT INCLUDED FOR EACH BLOCK

Figure E-1. Heads-Up Display Indicator Block Diagram

Developments indicate that volume and weight requirements will be reduced by a factor of two thirds using the new system of cockpit projection.

E.2 Advanced Display Devices

There is a continuing requirement for two-dimensional real-time video displays, exhibiting high brightness and high resolution and which require minimum power. The conventional electronically scanned cathode ray tube satisfies the intended requirements, but is undesirable for many applications because of depth requirements. An approach which gives great promise of success is the use of a matrix of discrete display elements with a matrix of control switches to convert a single channel video signal into a two-dimensional display pattern. The control must:

a. Be simple in design
b. Be capable of operating at high switching rates
c. Dissipate negligible power
d. Introduce negligible spurious signals
e. Be capable of accepting coincident row-column input signals

To date, most matrix display devices have employed the conventional electroluminescent capacitor as the display element. The performance of these systems was limited by the low brightness level and low efficiency of the display elements.

Single crystal injection electroluminescent diodes, although more efficient and brighter, are difficult and costly to fabricate on a matrix configuration.

A major drawback of the matrix display approach has been the difficulty of fabricating a large matrix of display elements and the attendant switching circuitry. Included in the fabrication problem is the difficulty of interconnecting the control circuits and combining the display matrix with the control circuits.

In view of the disadvantages of the electroluminescent capacitors and the single crystal injection electroluminescent diodes as matrix display elements, United Aircraft Corporation Research Laboratories have been considering the thin-film tunnel injection electroluminescent diode. Since this novel display element is produced by successive evaporations of insulators, and doped semiconductor compounds and metallic leads, it is predicted that large matrices can be fabricated by conventional masking approaches.

It is expected that resolutions greater than 10 to 20 lines per inch, and brightness levels greater than 15 foot-lamberts can be achieved with the thin-film tunnel diode. Diodes as small as 0.01 x 0.01 inch are within the state-of-the-art capabilities, thus providing 100-line per inch displays. Brightness levels beyond 100 foot-lamberts are also reasonable, with an upper limit of tens of thousands of foot-lamberts.

The characteristics of a sample array are summarized in Table E-1.

The possibility exists for the development of a sample display array with significantly more resolution than the 20 x 20 unit. The fabrication techniques (including advanced microminiaturization and interconnection) and the display element matrix construction methods discussed should provide the capability of constructing sample display arrays containing at least 10,000 resolution elements (100 x 100).

Table E-1. Sample Electroluminescent Array Characteristics

DISPLAY ELEMENT	
Type:	Thin-film tunnel injection electroluminescent diode (TELD)
	Pinlite incandescent bulbs
Form:	20 x 20 mosaic
Element Size	
Bulbs:	0.03 inch diameter bulbs
Diodes:	0.05 inch^2 (maximum)
Display Size:	2 inch x 2 inch (maximum)
Color	
Bulbs:	white
Diodes:	red
Brightness:	15 foot-lamberts (minimum)
Halftone Intensity Capability:	2 minimum (10 is an ultimate goal)
CONTROL CIRCUITRY	
Type:	20 x 20 row-column switching matrix
Sampling Period Per Element:	10 microseconds
Total Sampling Cycle Time:	4 milliseconds
Frame Rate:	250 cycles per second
Halftone Capability:	10 minimum
	SYSTEM
Power Dissipation:	7 watts

E.3 Quickening

The application of quickening techniques to the integrated display can contribute to improved pilot performance in the functions of monitoring the stability control system for possible malfunctioning, providing stabilization in the event of system failure, and maneuvering or maintaining a given attitude under manual control.

The quickening technique, as originally developed by Birmingham and Taylor (Section 15 , reference 9), employs summing the system output with its derivatives with appropriate gains in such a manner that the operator need only respond to a single indication of error. This technique, when properly applied, can improve the performance of the man-machine control system by providing the operator with an early knowledge of the eventual effects of his own control motions.

Quickening has been used successfully for many years in a contact analog display designed for submarine control in both vertical and horizontal planes. A simplified explanation of its operation is given in the following paragraphs.

Assume that the response to a transient disturbance is known to be of the form:

$$R(t) = R_1 - R_f)e^{-at} + R_f \qquad (1)$$

where:

$R(t)$ = response at time t

R_1 = initial output before disturbance

R_f = steady-state output after disturbance

For the purpose of quickening, it is desired to have an instantaneous estimate of R_f, the steady-state output after the disturbance. Since the only parameters available to the pilot for prediction and possible corrective action are the instantaneous system output $R(t)$ and its first time derivative $\dot{R}(t)$, we desire an expression for R_f in terms of $R(t)$ and $R(t)$. From equation (1), the first time derivative $\dot{R}(t)$ is given by:

$$\dot{R}(t) = -a(R_1 - R_f)e^{-at} \tag{2}$$

Combining equations (1) and (2) and rearranging terms, the desired equation for R_f is derived as:

$$R_f = R(t) + \frac{1}{a}\dot{R}(t) \tag{3}$$

If the desired steady-state output is R_o, the corrective action required by the pilot R_c is given by:

$$R_c(t) = R_f - R_o = R(t) + \frac{1}{a}\dot{R}(t) - R_o \tag{4}$$

The implementation of equation (3) in a pilot display is called a "quickened" display since it predicts, on a continuous basis, what the system output would be at some later time if no further corrective action was initiated.

Aside from the prediction aspect of quickening, the use of the technique has two other distinct advantages. One is that it can provide anticipation to counterbalance the effect of pilot lag in responding to rapidly varying outputs. This technique may, therefore, substantially alleviate the problem of the lateral instability experienced with the X-15 vehicle (Section 15 , reference 10). Since the pilot transfer function contains a form representing a pure lag of approximately 0.3 second (Section 15 , reference 11), and if the display could be made to represent an anticipation of approximately the same amount, pilot lag could be nearly eliminated. For example, if the display variable for sideslip angle β were modified to be:

$$\beta' = \beta + k\dot{\beta} \tag{5}$$

and β was assumed to be sinusoidal (i.e., $\beta = A \sin \omega t$), then $\dot{\beta}$, the first time derivative, would be given by:

$$\dot{\beta} = A\,\omega \cos \omega t \tag{6}$$

Combining the expressions for β and $\dot{\beta}$ into equation (5), we get:

$$\beta' = A\,[\sin \omega t + (k\omega) \cos \omega t]$$

or

$$\beta' = A \sqrt{1 + (k\omega)^2}\,\sin \omega\,(t + \frac{\theta}{\omega})$$

where

$$(\frac{\theta}{\omega}) = \frac{1}{\omega}\,[\tan^{-1}\,(k\omega)]$$

By expanding the inverse tangent function in a Taylor series for $|(k\omega)|^2 < 1$, we get

$$(\frac{\theta}{\omega} = k - \frac{1}{3} k^3\omega^2 + \frac{1}{5} k^5\omega^4 - \frac{1}{7} k^7\omega^5 + \ldots$$

Therefore, to a first order approximation, the effect of displaying β' instead of β is to add a pure lead of k seconds to the display to counteract the pure lag of the pilot. For a lead time of 0.3 seconds, the approximation of the series expansion is valid for frequencies up to one cycle per second.

The final advantage of a quickened display is that it simplifies the pilot's control task so that he may perform more functions with less training and less fatigue. This is substantiated by the results of the quickening study conducted at the Naval Research Laboratory in 1954 (Section 15 , reference 12).

E.4 Alphanumeric Indicators

Certain flight parameters (such as total altitude, actual heading, total velocity, and time) have no direct visual counterpart and are difficult to represent on an integrated display in a form allowing accurate interpretation. Therefore, where these parameters are required, either alphanumerics, bargraphs, or moving-tape type indications must be used. Any of these indications can be produced in a TV format. The accuracy of an alphanumeric readout is much greater than that of a bargraph, but the mechanization of the bargraph is far more simple and reliable, since it requires less than 1/20 of the circuitry (less than 1/7 of the size where state-of-the-art microelectronics are used to implement the alphanumerics). The moving-tape indicator allows accurate readout, but is much more complex to mechanize than a single alphanumeric.

For those flight parameters that do have adequate visual counterparts, some form of contact analog symbol or texture which is visually suggestive of the parameter in question should be used as the indicator. Where high accuracy readout is required, an alphanumeric can be positioned on the display in close positional association with the symbol which represents the parameter. Where such positioning is not required, a moving-tape method of alphanumeric indication can be provided on the CRT face. This method of quantitative readout results in higher display sensitivity without recourse to mechanical devices.

E.5 Advanced Reentry Display

A proposed format for an orbital-vehicle reentry display based on the requirements outlined in Section 15, reference 13 is shown in Figure E-2. Here, the basic X-15 format has been extended to include display of the following:

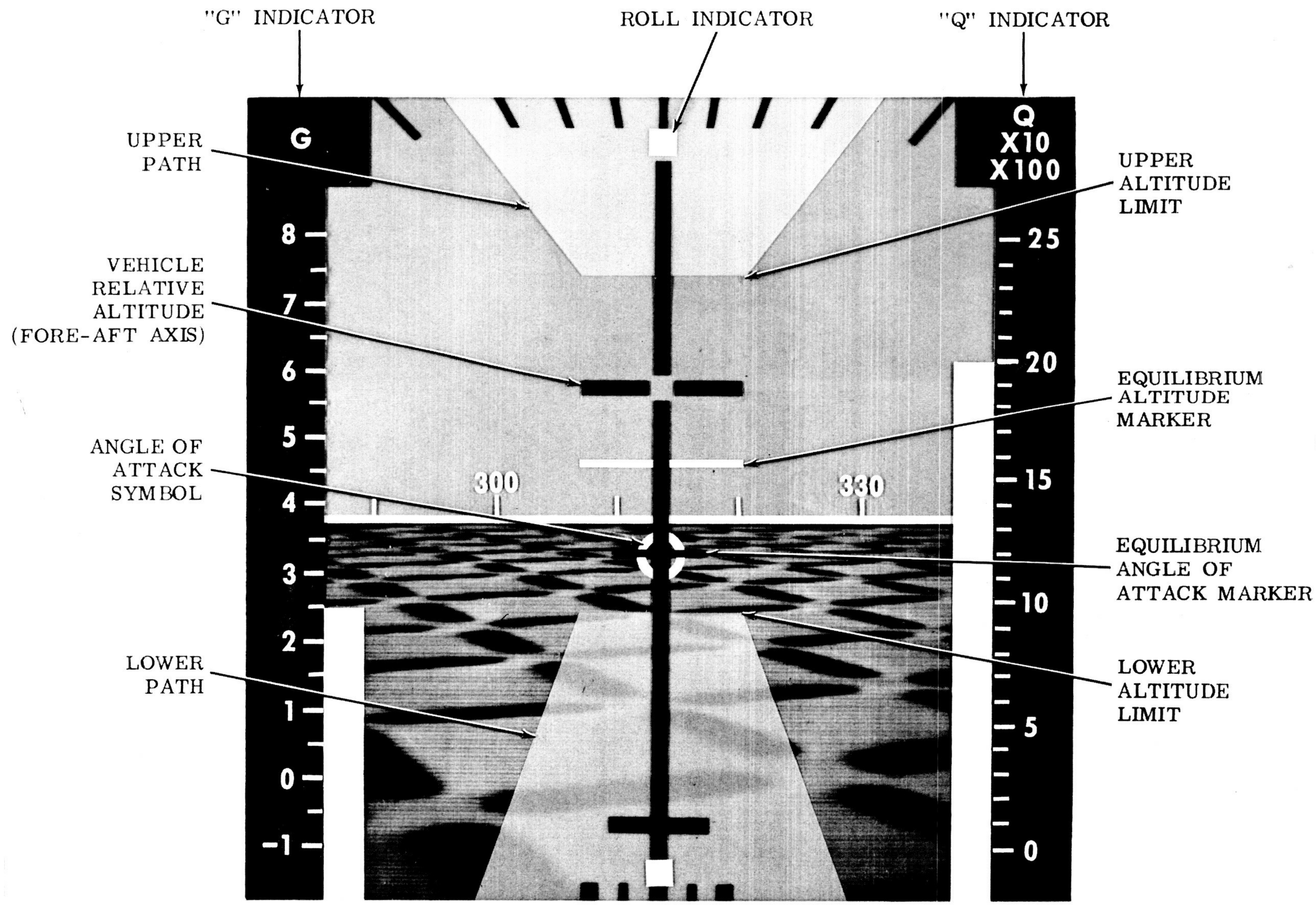


Figure E-2. Advanced Reentry Display

a. Upper and lower altitude limits
b. Relative vehicle altitude
c. Equilibrium - glide altitude and angle-of-attack
d. Quantitative indication of non-pictorial parameters (bargraphs)
e. Numeric readout of heading

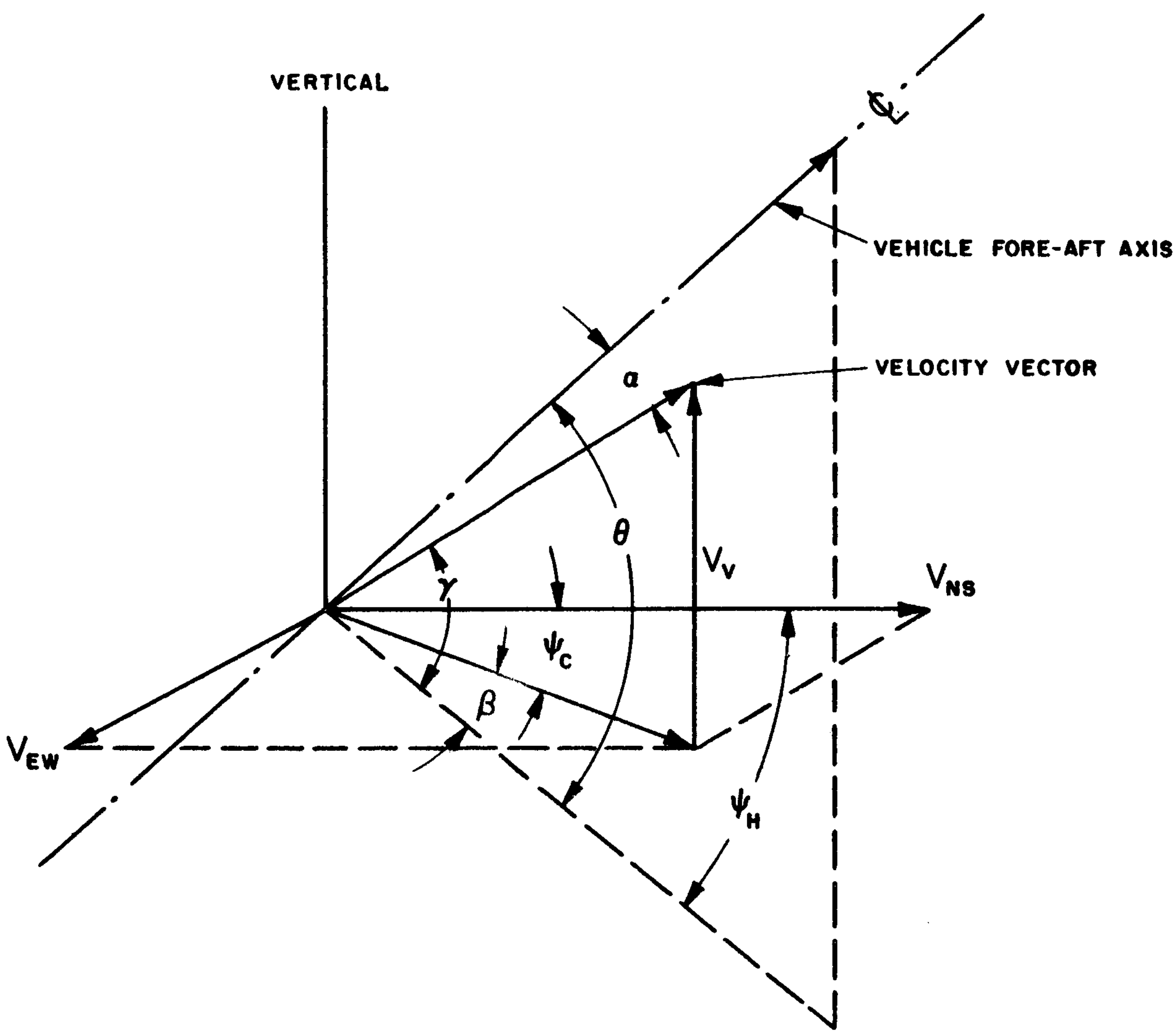
The altitude limits are displayed pictorially by upper and lower pathways which are shown cut off in a plane located at a fixed distance ahead of the aircraft. The vehicle position in relation to these limits is indicated by the vertical distances between the path cutoffs and the horizontal reticle line. For this indication to not change with pitch angle, the pathways are stabilized to the aircraft fore-aft axis. The horizon line provides the usual pitch reference. Angle-of-attack and sideslip are shown in the same manner as described in Section 8.

Quantitative indication of reentry acceleration levels and dynamic pressures is provided by video bargraphs on each side of the main display. Numerals and scales are generated by "scan-through" overlays, while actual values are readout by the relative position of a white vertical bar.

Numeric readout of heading is provided by numerals and heading reference markers along the horizon.

With a display of this type, necessary information is provided on one display for conducting critical reentry maneuvers, such as the one described, (i.e., change of altitude within the reentry corridor for range correction).

F. DEFINITION OF ANGLES



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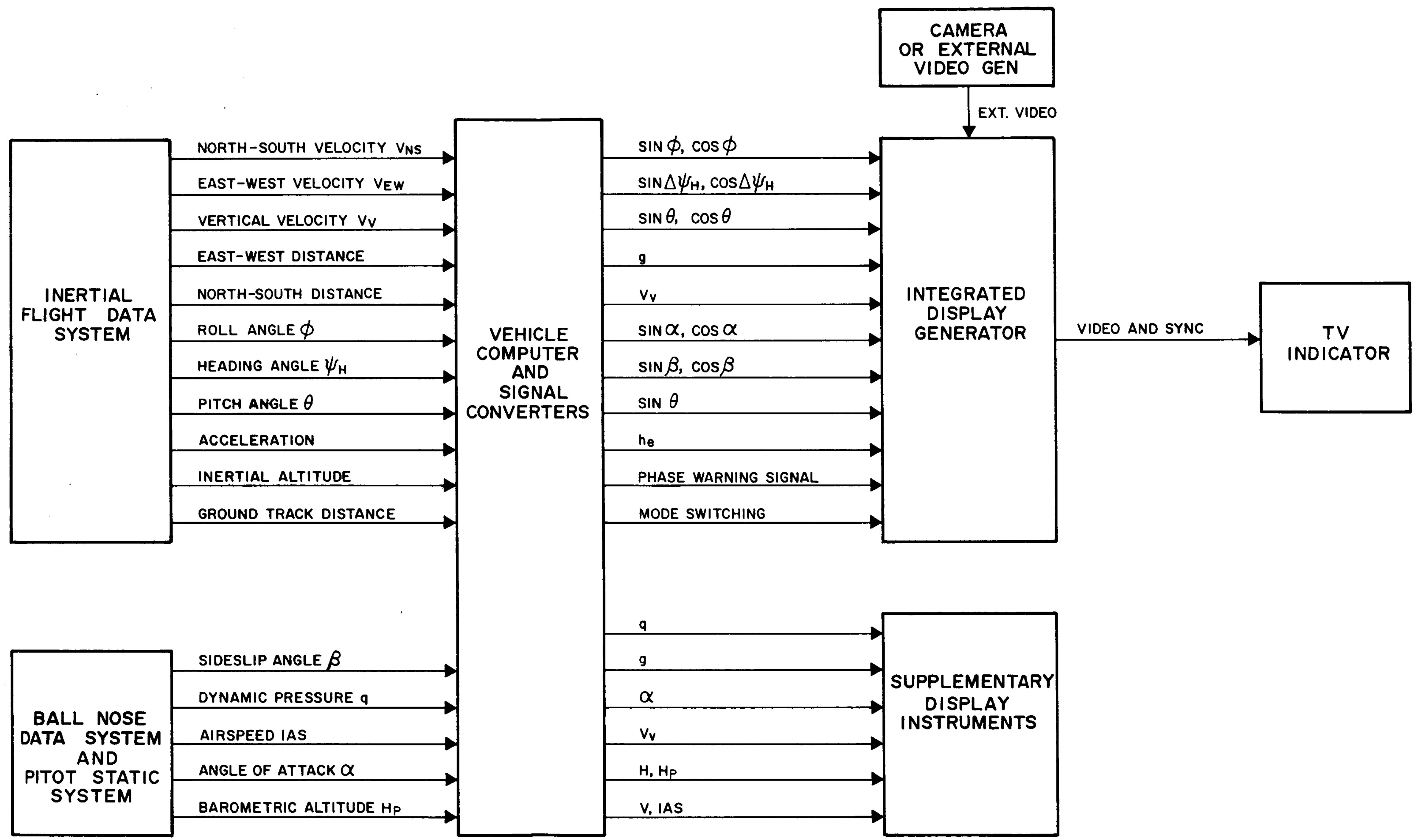

Figure 8-4. Input Block Diagram

PILOT TASK ANALYSIS - ALTITUDE MISSION

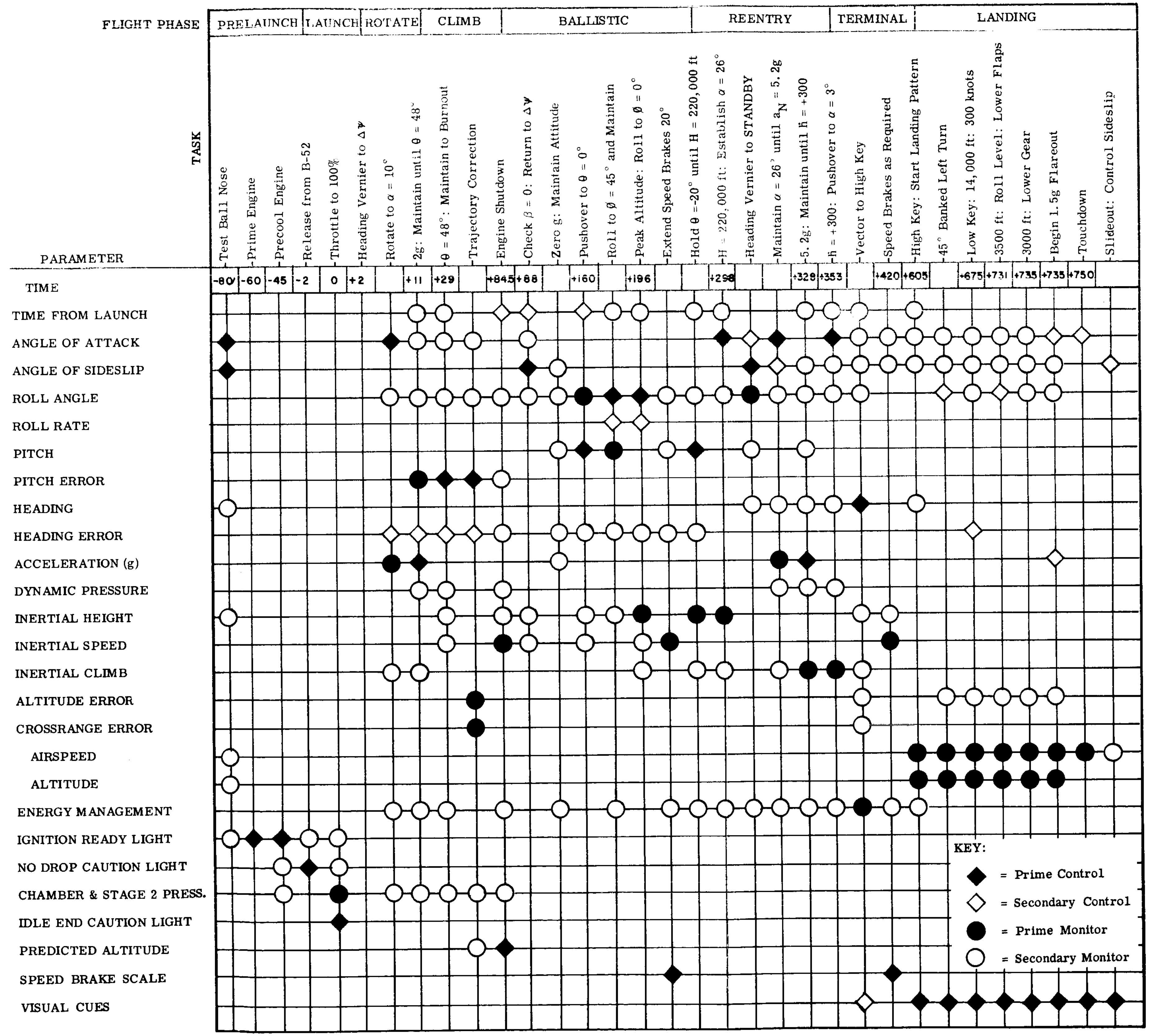

Figure 9-2. Pilot Task Analysis.

Table 7-1. Instrument Characteristics

INDICATOR	FIGURE	RANGE	SCALE SIZE	MINIMUM DIVISION	NUMBERING INCREMENT	INDICATED ACCURACY	VIEWING SENSITIVITY	PANEL AREA	DEPTH	WEIGHT	POWER	SENSOR
Angle-of-Attack	7-3a	-10 to +40 degrees	2.7 inches	1 degree	10 degrees	± 1/4 degree	18.5 degrees per inch	3-inch diameter	7 inches	31 pounds	10 VA	Ball Nose
Angle-of-Sideslip	7-2	-15 to +15 degrees	2 inches	Null	Null	± 1 degree	17 degrees per inch	Note 1	Note 1	Note 1	Note 1	Note 1
Accelerometer	7-3b	-5 g to +12 g	2.7-inch diameter	0.2 g	2 g	± 0.2 g	2 g per inch	3-inch diameter	3-1/2 inches	1.51 pounds	None	Self-Contained
Airspeed	7-1	80-850 knots	1.7-inch diameter	10 knots	100 knots	± 5 knots	See Paragraph 7.7	3-inch diameter	4-1/2 inches	2 pounds	None	Pitot and Static Pressure
Mach		.4 to 2.6 Mach	2.7-inch diameter	0.05 Mach	0.1 Mach	± 0.005 Mach						
Pressure Altimeter	7-1	-1000 to +80,000 feet	2.7-inch diameter	20 feet	100 feet	± 280 feet maximum	150 feet per inch	3-inch diameter	4-1/2 inches	2 pounds	None	Pitot and Static Pressure
Dynamic Pressure	7-3c	2500 pounds per square foot	2.7-inch diameter	50 pounds per square foot	500 pounds per square foot	± 3 percent (Note 2)	400 pounds per square foot per inch	3-inch diameter	3-1/2 inches	2 pounds	None	Ball Nose
Inertial Height	7-4a	10^6 feet	2.7-inch diameter	2000 feet	10,000 feet	± 5000 feet (Note 3)	15,000 feet per inch	3-inch diameter	9 inches	1.51 pounds	15 VA	Inertial Platform
Inertial Speed	7-4b	7000 feet per second	2.7-inch diameter	2000 feet per second	1000 feet per second	± 200 feet per second	1000 feet per second per inch	3-inch diameter	9 inches	2 pounds	15 VA	Inertial Platform
Inertial Climb	7-4c	± 1000 feet per second	2.5 inches	100 feet per second	200 feet per second	± 100 feet per second	1000 feet per second per inch	3-inch diameter	3-1/2 inches	0.5 pound	15 VA	Inertial Platform
Roll Rate	7-1	± 120 degrees per second	2.7-inch diameter	10 degrees per second	50 degrees per second	5 degrees per second	75 degrees per second per inch	3-inch diameter	7 inches	3 pounds	10 VA	Self-Contained
Stopwatch	7-1	3000 seconds	2-inch diameter	1 second	10 seconds	0.1 second	16 seconds per inch	2.5-inch diameter	---	---	None	None

NOTES

1. Refer to Attitude Indicator Characteristics
2. Error increases below Mach 2.5
3. Typical after 10 minutes